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CLUSTERED LANCE BOOSTER (CLB)

FEASIBILITY STUDY

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CLUSTERED LANCE BOOSTER FEASIBILITY STUDY

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Foreward

This feasibility study was performed by Loral Vought (LV) Systems Corporation (formerly LTV Aerospace and Defense Company, Missiles Division) for the U.S. Army Strategic Defense Command, Huntsville, Alabama, under Contract DASG60-92-C-0120.

The work was performed at the LV facilities in Grand Prairie, Texas, between 29 September 1992 and 28 January 1993.

Summary

LV was requested to investigate the feasibility of clustering four Lance propulsion systems as the first stage of a two-stage target vehicle. Two vehicle configurations were considered, both using the Clustered Lance Booster (CLB) as the first stage. On one configuration, the second stage will be the 40-inch diameter Liquid Fueled Target (LFT-40), which was the subject of an LTV Aerospace and Defense Company study in 1991. This configuration is designated as the CLB/LFT-40. The second stage of the other configuration will be a modified Lance having the same external shape as the Lance missile (22-inch diameter). This configuration is designated as the CLB/LFT-22.

Detailed analyses were performed on the CLB/LFT-40 configuration to determine its technical feasibility and performance. The technical feasibility and performance evaluation of the CLB/LFT-22 was then based on its similarity to the CLB/LFT-40.

The contractual statement of work is included in this report as appendix A. A summary of the work performed is as follows:

A. Conceptual design definition
B. Preliminary aerodynamic loads analysis
C. Preliminary structural loads analysis
D. Structural, stress and weights analysis
E. Required control authority
F. Launch rail requirements
G. Flight performance projections
H. Additional studies required before initiation of full scale development.

The analyses were performed in sufficient detail to determine technical feasibility and to identify areas which need additional study. No attempt was made to optimize the design.

Of particular concern was the control authority required to offset the forces and moments produced by differences in thrust magnitude and alignment of the four propulsion systems which make

up the CLB. It was found that the Lance thrust vector control system provides adequate control after about 0.4 second of thrust. Control during the first 0.4 second can be achieved by designing the launcher so that the vehicle does not reach the end of the rail for 0.4 second. This would require hold-back for a short period of time and a rail of appropriate length. For example, for a hold-back time of 0.2 second, the required rail length is six feet for the CLB/LFT-40 vehicle. The requirement for hold-back time and launch rail length could be relaxed by installing a hot gas reaction control system in the upper stage. A conceptual design for such a system is included in this study.

Loads and stress analysis indicated that the fin attachment points on the Lance tank body are inadequate for the CLB application. It will be necessary to provide additional structure to support the fins. This will drive vehicle weight above the baseline estimates given in section 2.4 of this report.

Additional information, such as the following, will be required to conduct the follow-on studies:

A. Mission requirements for the CLB vehicle
B. Mass properties, structural characteristics, aerodynamic characteristics and propulsion system performance for the second stage
C. Definition of the interface between the two stages

Conclusions

The study indicates that it is technically feasible to cluster four Lance propulsion systems to form the CLB, but that there are significant problem areas that require additional study before full-scale development is initiated.

Recommendations

It is recommended that funding be provided to conduct the follow-on studies outlined in appendix B of this report.

1. BACKGROUND

The U.S. Army Strategic Defense Command has a requirement for a liquid propellant booster to serve as the first stage of a target vehicle. The Lance propulsion system (which is a liquid propellant system) has been proposed for this application because it is currently being decommissioned by the U.S. Army and is available in quantity. Four Lance propulsion systems would be clustered to form the first stage. The feasibility of developing the Clustered Lance Booster (CLB) is the subject of this report.

A detailed description of the Lance missile and its tactical operation are given in appendix B. The Lance propulsion system operation in the CLB would differ from its tactical operation in the following ways:

A. Both the booster engine and sustainer engine will operate until propellant depletion.
B. The spin torque system will be removed.
C. The Lance guidance and control system will be removed. Directional control signals to the thrust vector control (TVC) valves will be provided by the target vehicle guidance and control system.

Prior to launch of the CLB it will be necessary to remove the antipropulsion unit (APU) and to arm the igniter. The propulsion systems must be oriented in the CLB in such a manner that access to these devices is available.

The Lance propulsion system has a hot gas relief valve (HGRV) which vents excess gas from the solid propellant gas generator (SPGG) through diametrically oppose vent tubes. The propulsion systems must be oriented in the CLB in such a manner that the SPGG gases are not vented into the space between the propulsion systems.

2. TECHNICAL DISCUSSION

2.1 Conceptual Design

The baseline design for the CLB is shown in figures 2.1-1 through 2.1-3. This is the design which was used in the analyses which follow. It is referred to as "the baseline design". Loads and stress analyses performed after the baseline design was completed indicated that strengthening of the structure will be required. The areas requiring modification are indicated in section 2.3.2.

The conceptual design baseline is predicated on the requirement that Lance be assembled into a cluster at the launch complex in its fueled, ready-to-fire condition. This prerequisite presents a design constraint which limits attachment of the cluster and interstage to clamped and/or bolted assemblies.

The four tank body assemblies require specific orientation within the cluster to prevent impingement of the vented SPGG gasses and also to allow access to the Safe and Arm (S&A) Device and the APU's on each of the tanks. A spacing of one-half inch between each of the tanks is also required to allow for tank expansion during boost and additionally to allow for the protrusion of the external tunnel.

The proposed design utilizes the warhead mounting interface of the tanks for attachment of a plate which ties together the forward end of the cluster and also provides a mid-ring for attachment of the second stage. Each tank is bolted and pinned to the plate providing for rigid attachment of the tank assemblies.

The aft end of the tanks are tied together at the rear bulkhead. Three of the four existing fin post holes on each tank are tapped to accept a one and one-fourth inch threaded member which is used to attach brackets to the tanks. A tube section is bolted at the center of the cluster which provides the required spacing and in conjunction with the outer brackets, secures the tanks to each other.

The mounting plate described above is also used to attach the second stage to the cluster. The second stage is clamped to this mid-ring using a vee-groove clamping device. Separation of stages is achieved through the use of explosive bolt cutters in the clamp.

An aluminum sheet metal fairing is provided for aerodynamic flow prior to separation of the stages. The fairing is attached to both the cluster and second stage ends with vee clamps utilizing explosive bolt cutting devices for separation. Linear shaped charges will be used for separation

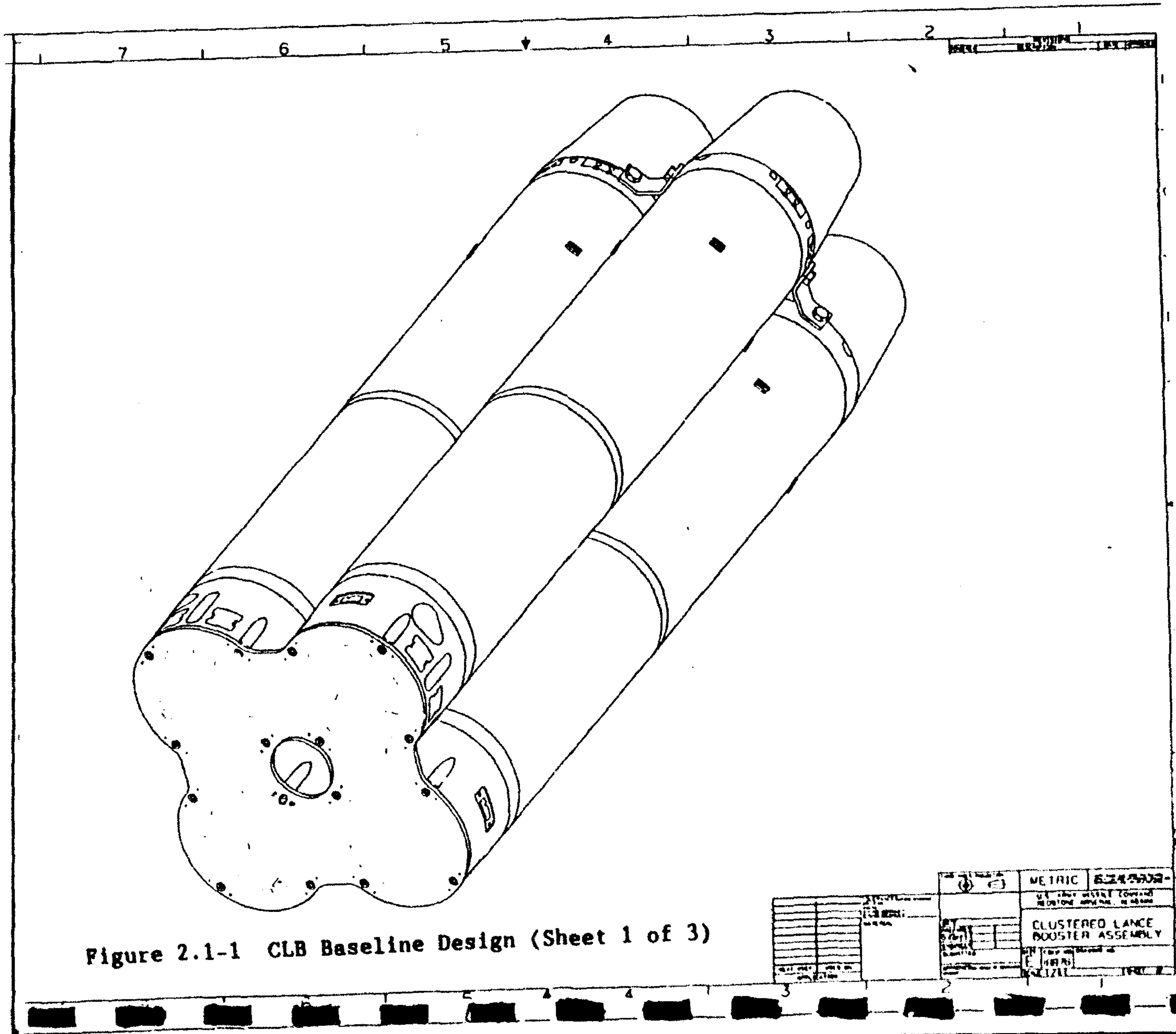


Figure 2.1-1 CLB Baseline Design (Sheet 1 of 3)

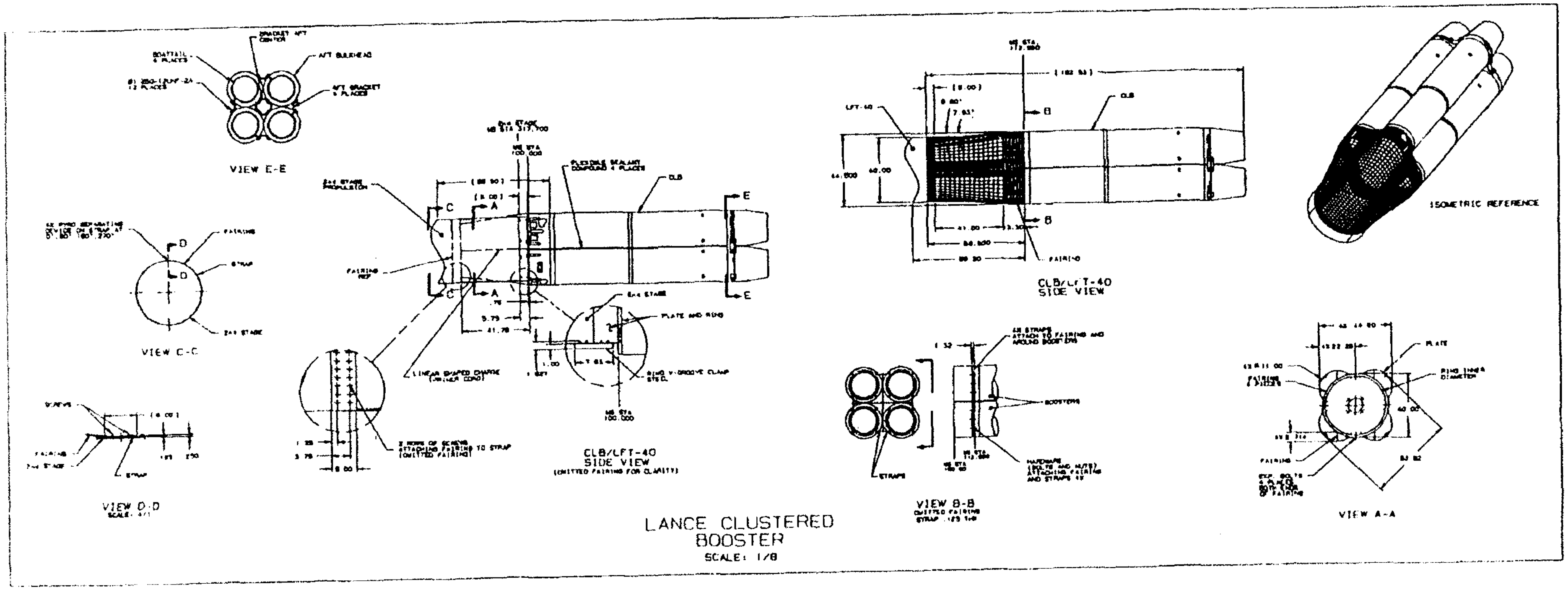


Figure 2.1-2 CLB Baseline Design (Sheet 2 of 3)

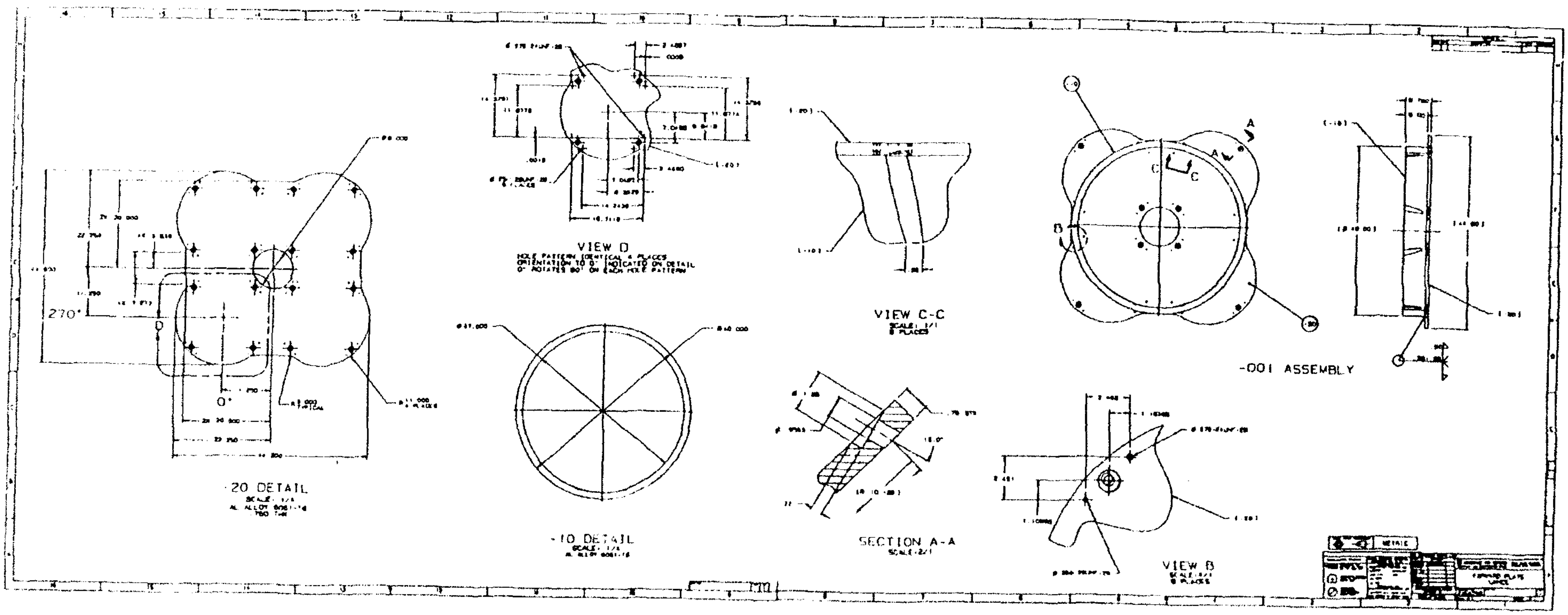


Figure 2.1-3 CLB Baseline Design (Sheet 3 of 3)

of the sheet metal skin of the fairing.

2.2 Flight Dynamics and Aerodynamics

As part of the study to determine feasibility of the concept, preliminary aerodynamic predictions have been made for the total CLB vehicle. Stability characteristics of configurations with fins were estimated and used as input to flight control analysis work. Aerodynamic data predictions were also necessary to support flight trajectory simulation. Predictions of the aerodynamic load distribution of the vehicle were provided to structural engineers for flight loads analysis. Estimates of the airloads on the fins were also provided.

2.2.1 Methodology

Two aerodynamic prediction tools were used for the preliminary study. The following paragraphs describe the codes used and the limitations of each.

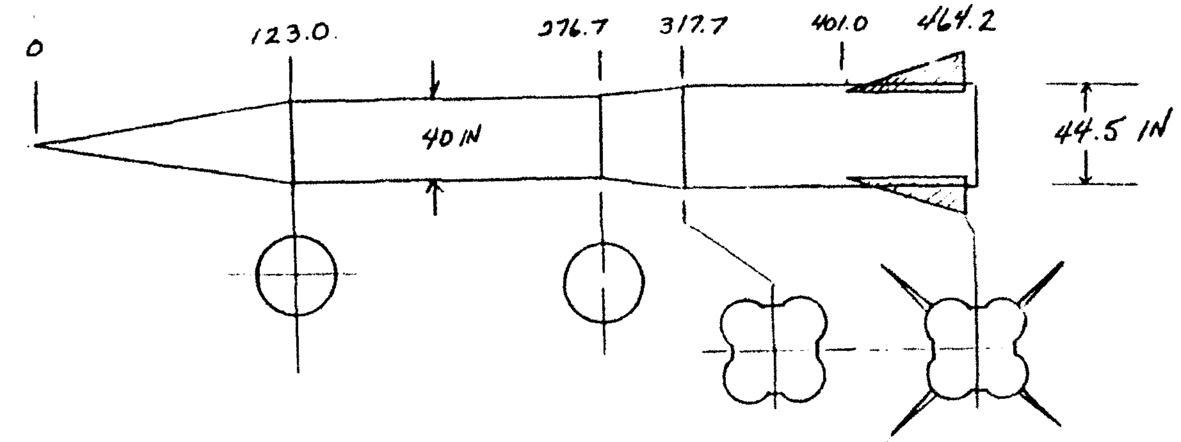
The Naval Surface Weapons Center aerodynamic prediction program, known as MOORE at Loral Vought Systems, was used for subsonic Mach number conditions (reference 2.2-1). MOORE is limited to basic configurational geometry input. The configuration body must be circular in cross-section and may include fins of various size and shape. MOORE uses a semi-empirical prediction method for subsonic calculations. Results from MOORE have approximately a ninety percent confidence level.

Supersonic aerodynamic predictions were made using the Zonal Euler Solver (ZEUS) code. ZEUS is a supersonic space marching Euler Solver capable of predicting external flow for various missile-type configurations. An advantage for using ZEUS for the CLB predictions is its capability of predicting aerodynamics for non-circular cross-sections. ZEUS predictions are limited for conditions which approach Mach numbers of one because of the possibility of subsonic flow. The confidence level for ZEUS results is approximately ninety-five percent.

2.2.2 Code Usage

The configurational geometry used as input for each code is shown in figure 2.2-1. The ZEUS model consists of a conical nose, cylindrical mid-body, a flared section, and an aftbody section with a cross-sectional shape which outlines the clustered boosters. The base of the flared section conforms to the shape of the clustered booster outline.

ZEUS PREDICTION MODEL (SUPERSONIC MACH RANGE)



MOORE PREDICTION MODEL (SUBSONIC MACH RANGE)

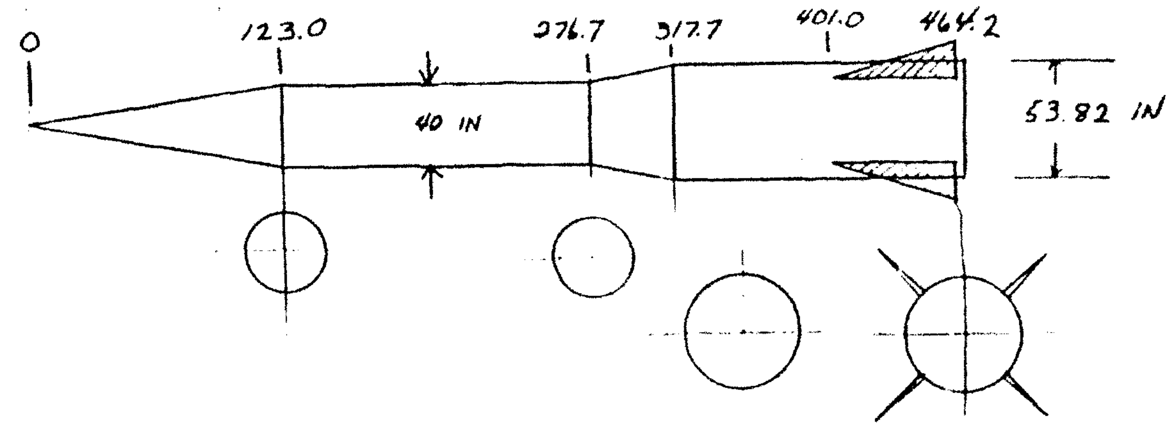


Figure 2.2-1 Geometric Models for Aerodynamic Predictions

The MOORE model consists of body components with circular cross-sections only. An assumption was made that a circular cross-section geometry with a radius equal to the maximum radius of the clustered booster shape would produce similar aerodynamic characteristics. Comparisons of MOORE supersonic data with ZEUS supersonic data verify that this is a reasonable assumption.

These models were created with the assumption that material would be placed in between the clustered boosters so as to prevent air flow between the boosters.

In order to do fin sizing parametrics, sets of fins were added to the body geometry of both the MOORE and ZEUS models. Figure 2.2-2 shows the four fin sizes used in the predictions.

2.2.3 Aerodynamic Prediction Results

Aerodynamic predictions were made for the CLB configuration covering the flight conditions of the simulated trajectory.

Figures 2.2-3 and 2.2-4 show the variation of normal force and center of pressure with Mach number for the different fin sizes. Fin sizing results indicate that in order to have adequate stability for control, the fin needs to have at least 1.75 times the area of the original Lance fin.

Figures 2.2-5 and 2.2-6 give the normal force coefficient derivative and the pitching moment coefficient derivative for the CLB configuration with 1.75 area delta fins. Moment data is given for both a launch condition and burn-out condition.

The zero angle of attack drag coefficient predictions are given in figure 2.2-7 for the complete configuration and in figure 2.2-8 for the second stage configuration. The drag data was calculated for sea level altitude. The two curves on each plot show the difference in drag due to base pressure.

2.2.4 Aerodynamic Load Distribution

The aerodynamic load distribution was predicted for the CLB configuration at the maximum load flight condition. The maximum aerodynamic load occurs approximately at maximum dynamic pressure. For the CLB trajectory maximum dynamic pressure is about 86.5 psi and occurs at a Mach number of 3.4. The normal force coefficient distribution at maximum aerodynamic load is plotted versus distance from the nose in figure 2.2-9. This distribution includes the carry-over load of the fins onto the body. The aerodynamic load of the tail fins is shown as a point load at approximately two-thirds the fin root chord length.

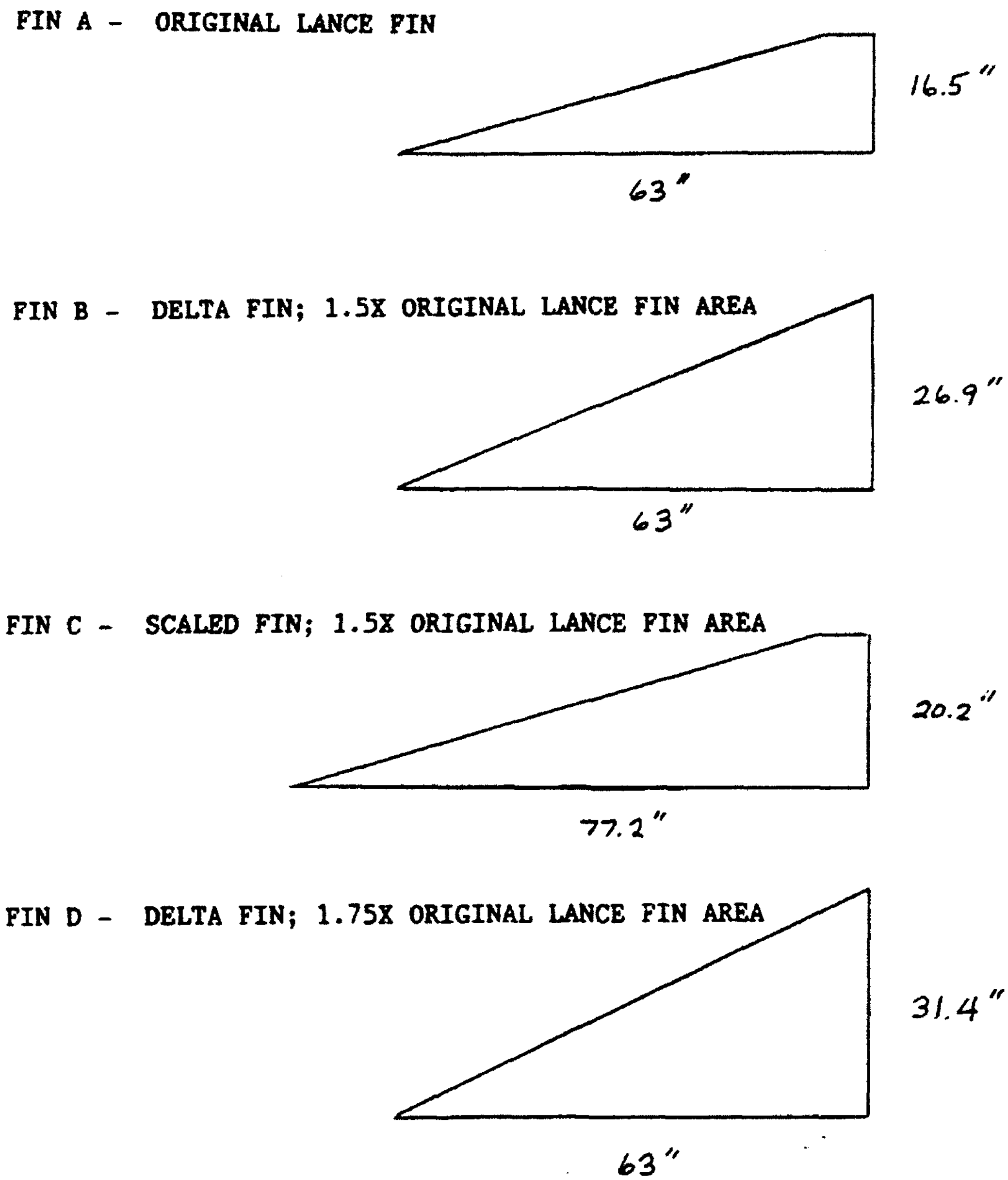


Figure 2.2-2 Fin Geometry Used in Aerodynamic Predictions

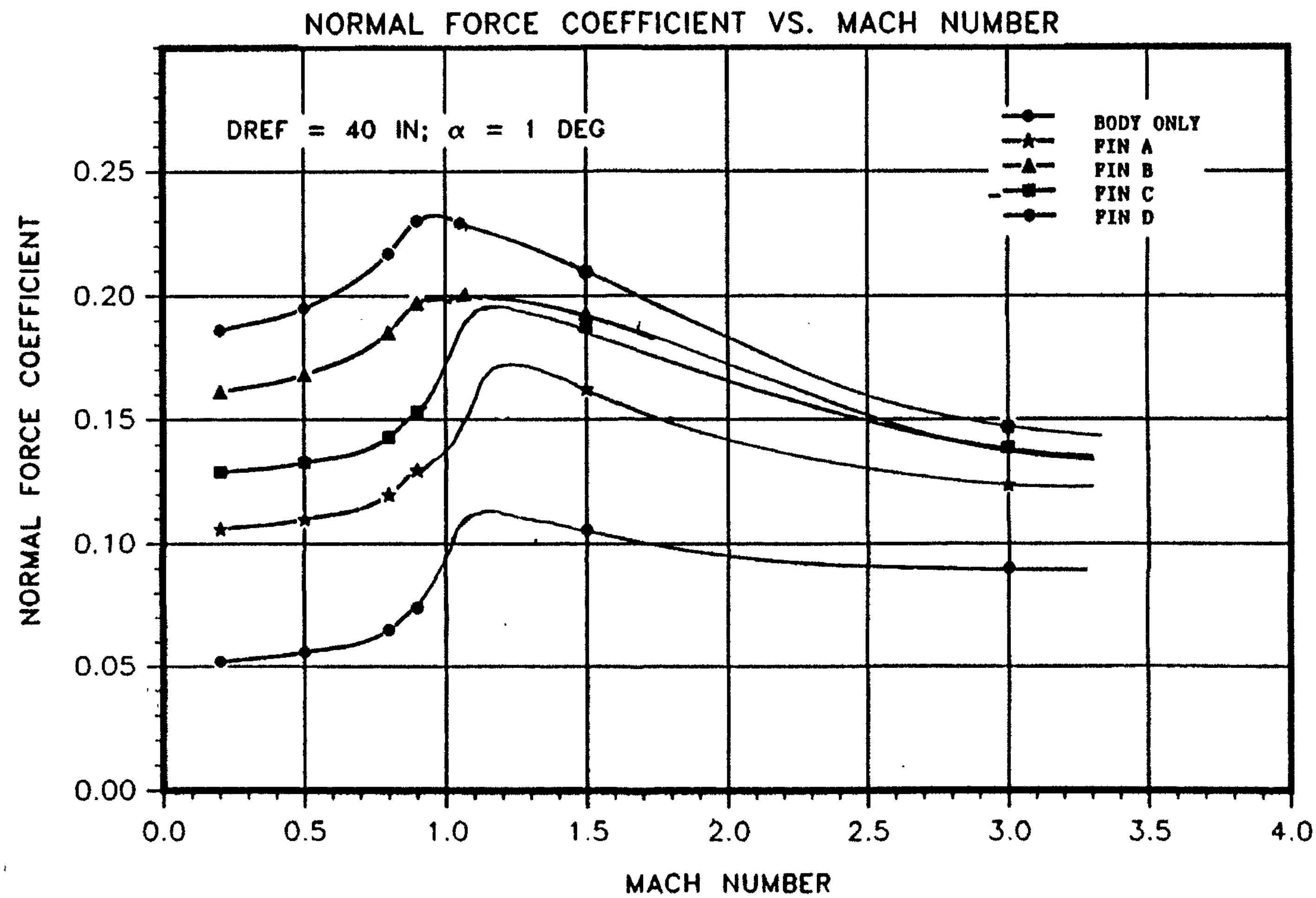


Figure 2.2-3 Normal Force Coefficient vs. Mach Number: Different Fin Sizes

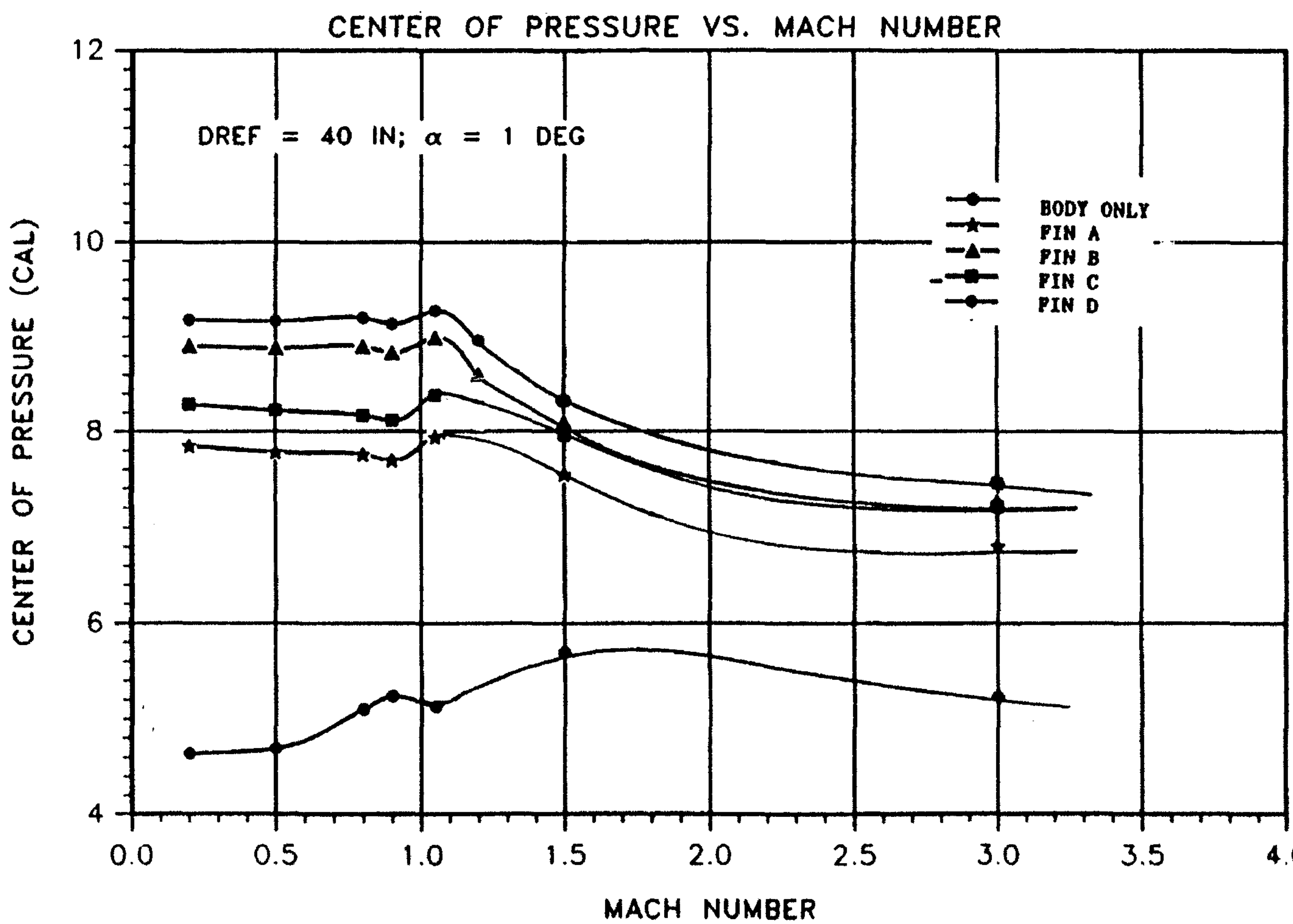


Figure 2.2-4 Center of Pressure vs. Mach Number: Different Fin Sizes

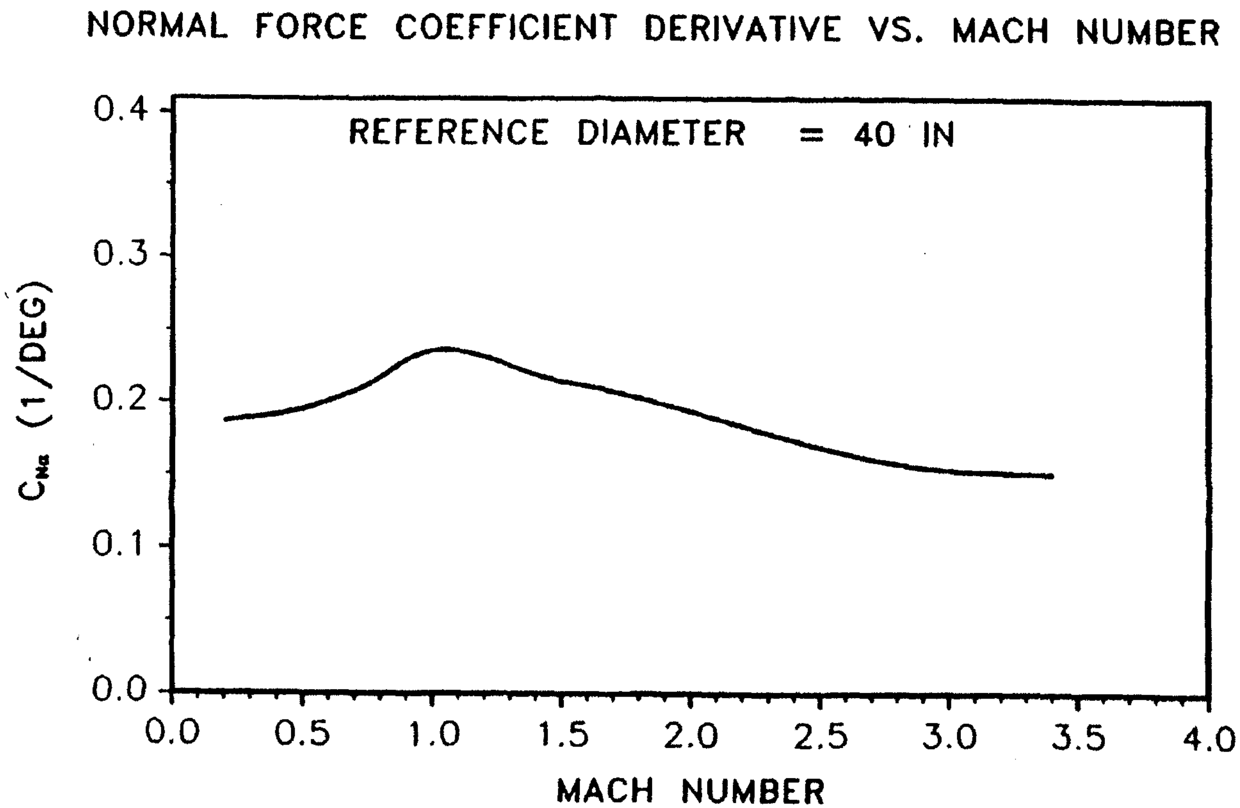


Figure 2.2-5 Normal Force Coefficient Derivative vs. Mach Number: Fins are 1.75 times area of original Lance fins

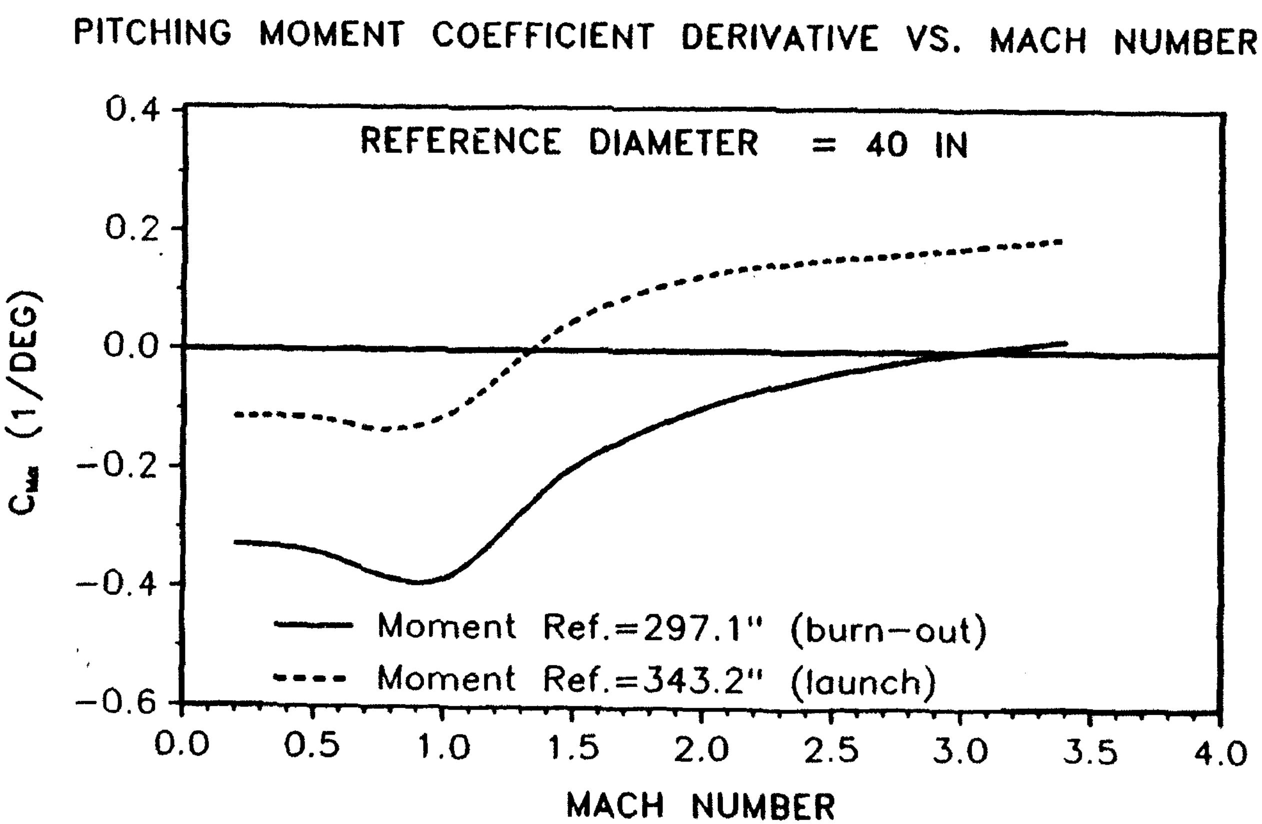


Figure 2.2-6 Pitching Moment Coefficient Derivative vs. Mach Number: Fins are 1.75 times area of original Lance fins

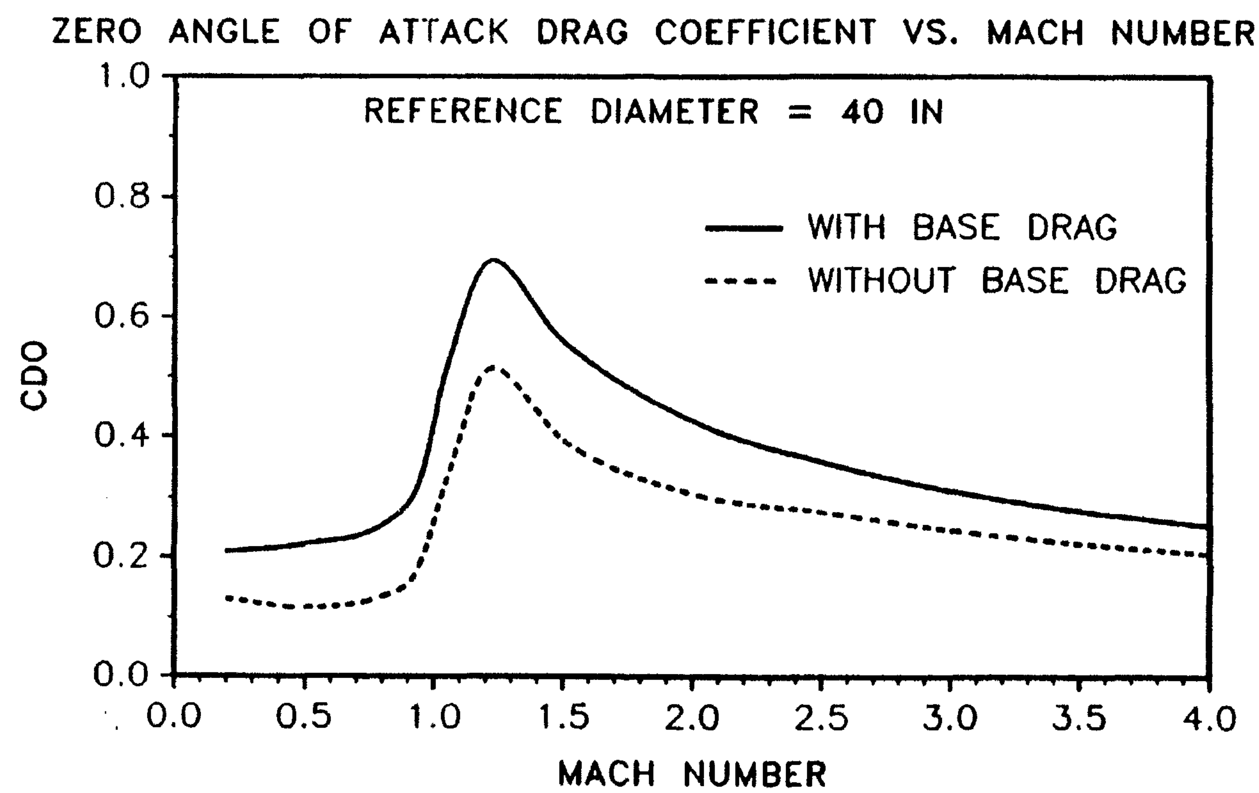


Figure 2.2-7 Zero Angle of Attack Drag Coefficient vs. Mach Number; Total Configuration

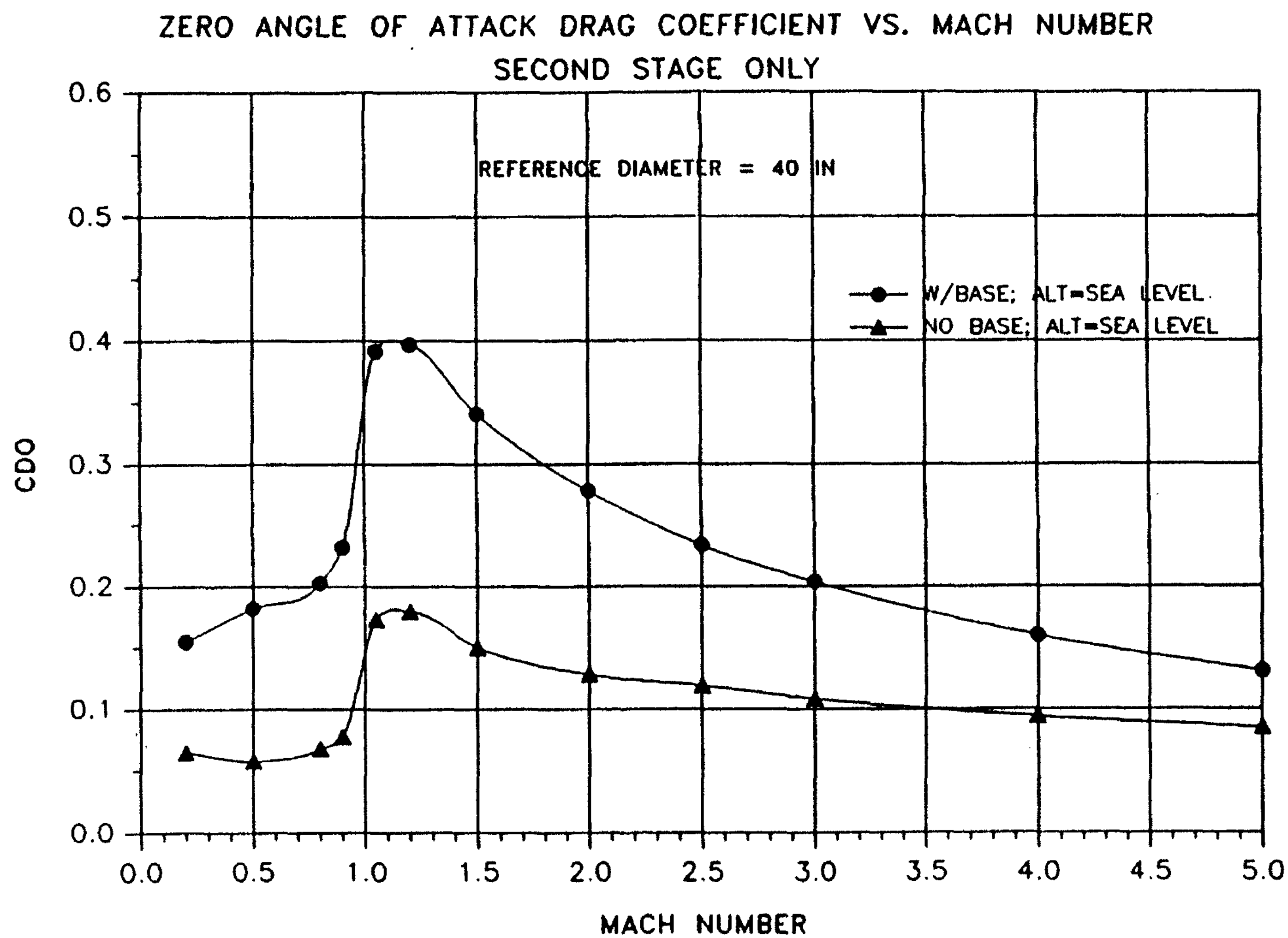


Figure 2.2-8 Zero Angle of Attack Drag Coefficient vs. Mach Number; Second Stage Configuration

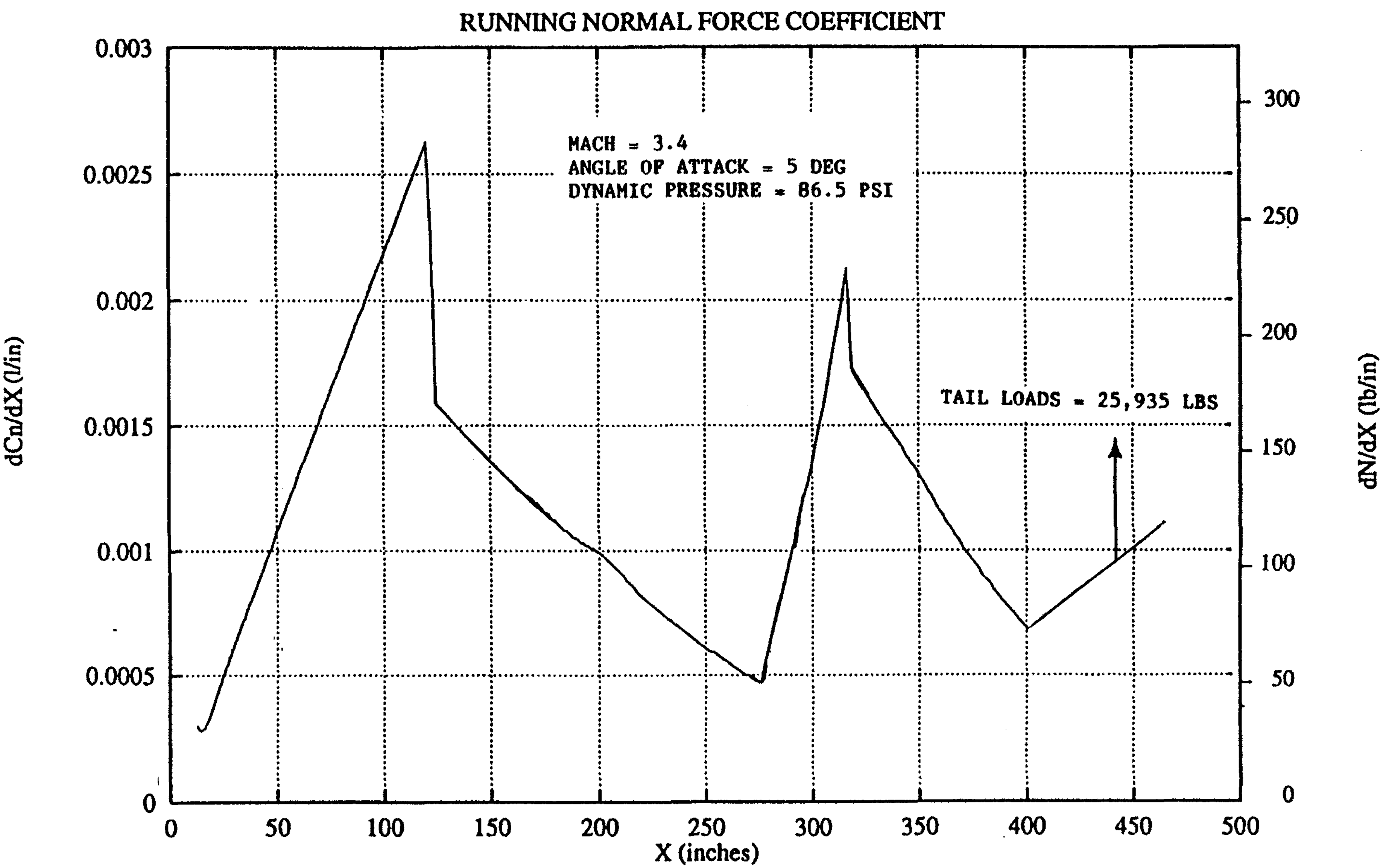


Figure 2.2-9 Body Aerodynamic Load Distribution

2.3 Structural Design and Analysis

2.3.1 Flight Loads

The flight condition which, in general, imposed the most critical structural loading on the vehicle occurred at the maximum dynamic pressure (q) flight condition. This condition occurred at Mach number 3.0 (q= 86.5 psi) with 20.8 g axial acceleration. An angle-of-attack of 5.0 degrees was assumed. The running aerodynamic loads for this condition are shown in figure 2.3-1. These are combined with the vehicle mass distribution (shown in figure 2.3-2) for corresponding flight condition. A rigid body vehicle simulation was assumed to obtain maximum axial loads, shear loads and bending moments at various stations. Based on past experience, these loads resulting from the rigid body type analysis were scaled up a factor of 1.15 to account for vehicle flexibility.

The predicted axial load, shear load, and bending moment distributions for CLB configuration imposed maximum q condition are summarized in figures 2.3-3 through 2.3-5. The axial loads at maximum thrust condition (axial acceleration = 21.1 g) are included in figure 2.3-3. The maximum bending moment is 1.759E6 in-lb which occurred at missile station 250 inch.

2.3.2 Lance Structural Limitations

The use of existing LANCE motors as a propulsion system for an interceptor vehicle using a clustered arrangement of LANCE motors is a challenging concept and appears to be feasible if the design is properly conceived recognizing the capabilities of a LANCE motor case as a structural member in the more highly loaded CLB configuration. The following paragraphs present some of the structural limitations of the LANCE motor and how a practical airframe structure can be designed to minimize these inherent limitations.

2.3.2.1 Lance Motor Case Limitations

The original LANCE motor cases were designed to handle light loads compared to the CLB configuration. In particular, the CLB will be required to handle fin loads approximately ten times larger than LANCE. The redistribution of each motor thrust load must be applied to the payload section through the existing motor guidance and control aluminum casting by means of a different load path than used on LANCE into a redistribution structure at the front end of the motor. The over-riding limitation on using the motors as primary structure is that they are initially fueled and cannot be modified.

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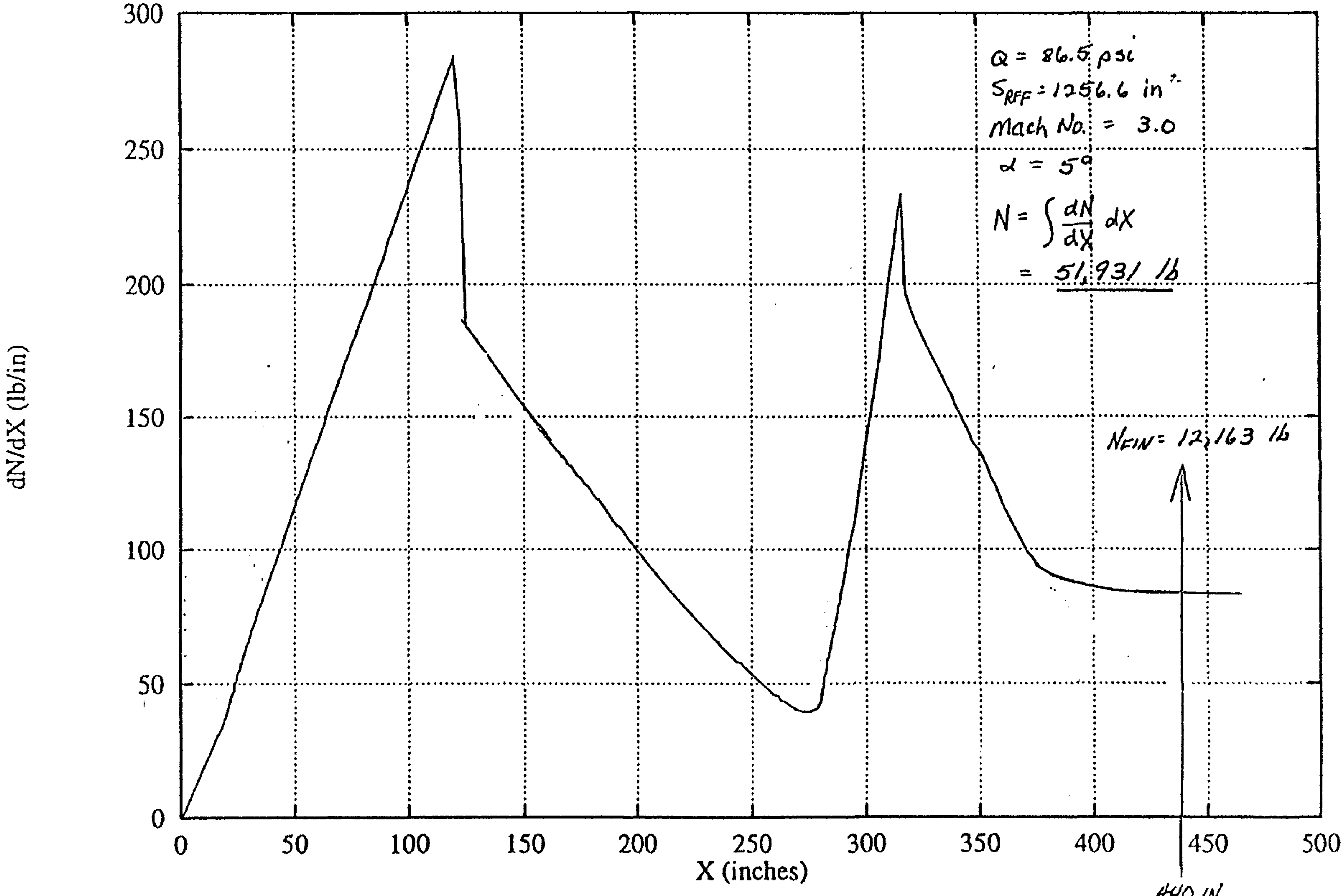

Figure 2.3-1 Running Normal Force

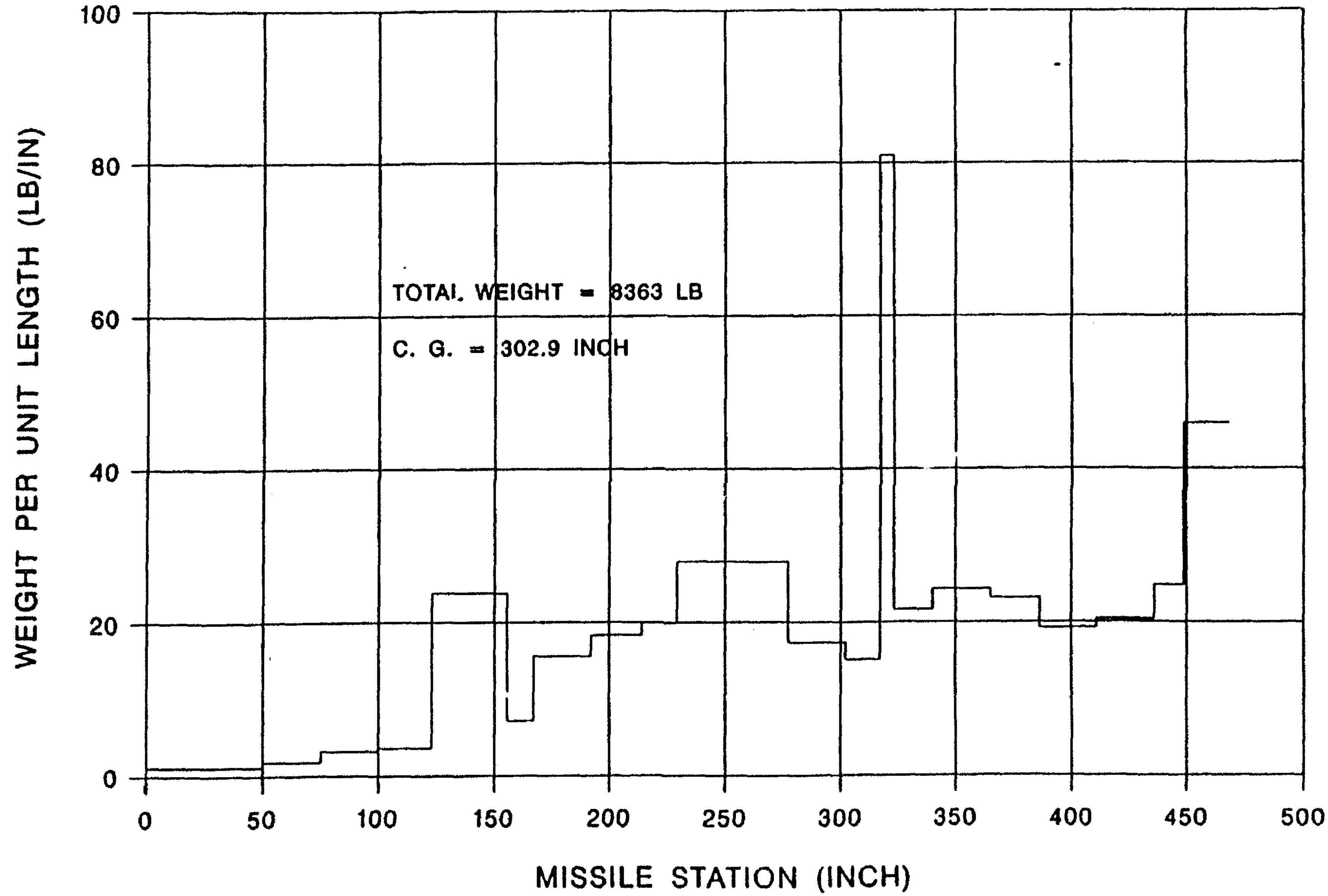

Figure 2.3-2 Weight Dictribution of CLB Missile at Maximum Q Flight Condition

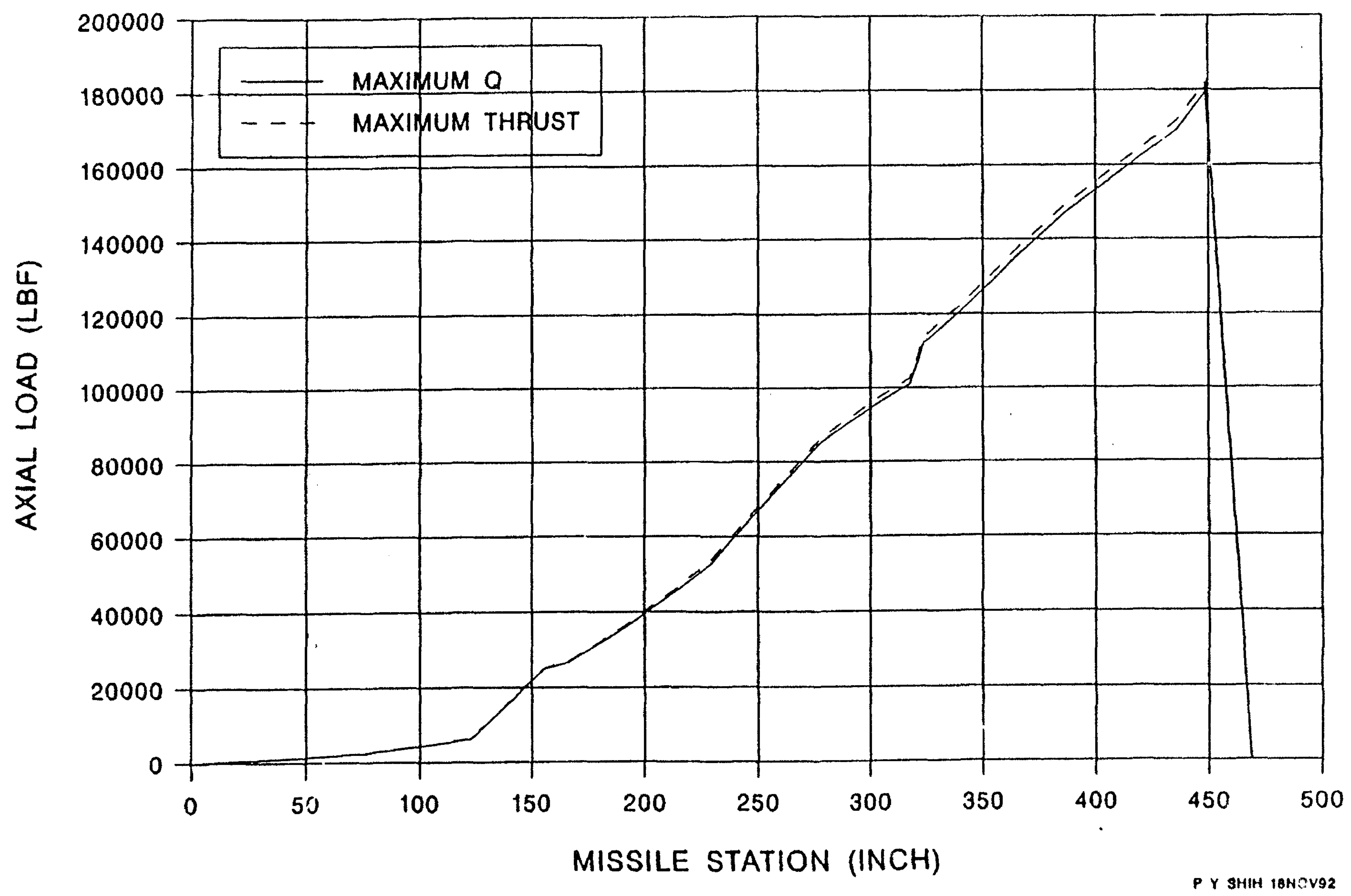

Figure 2.3-3 Axial Loads of CLB Missile

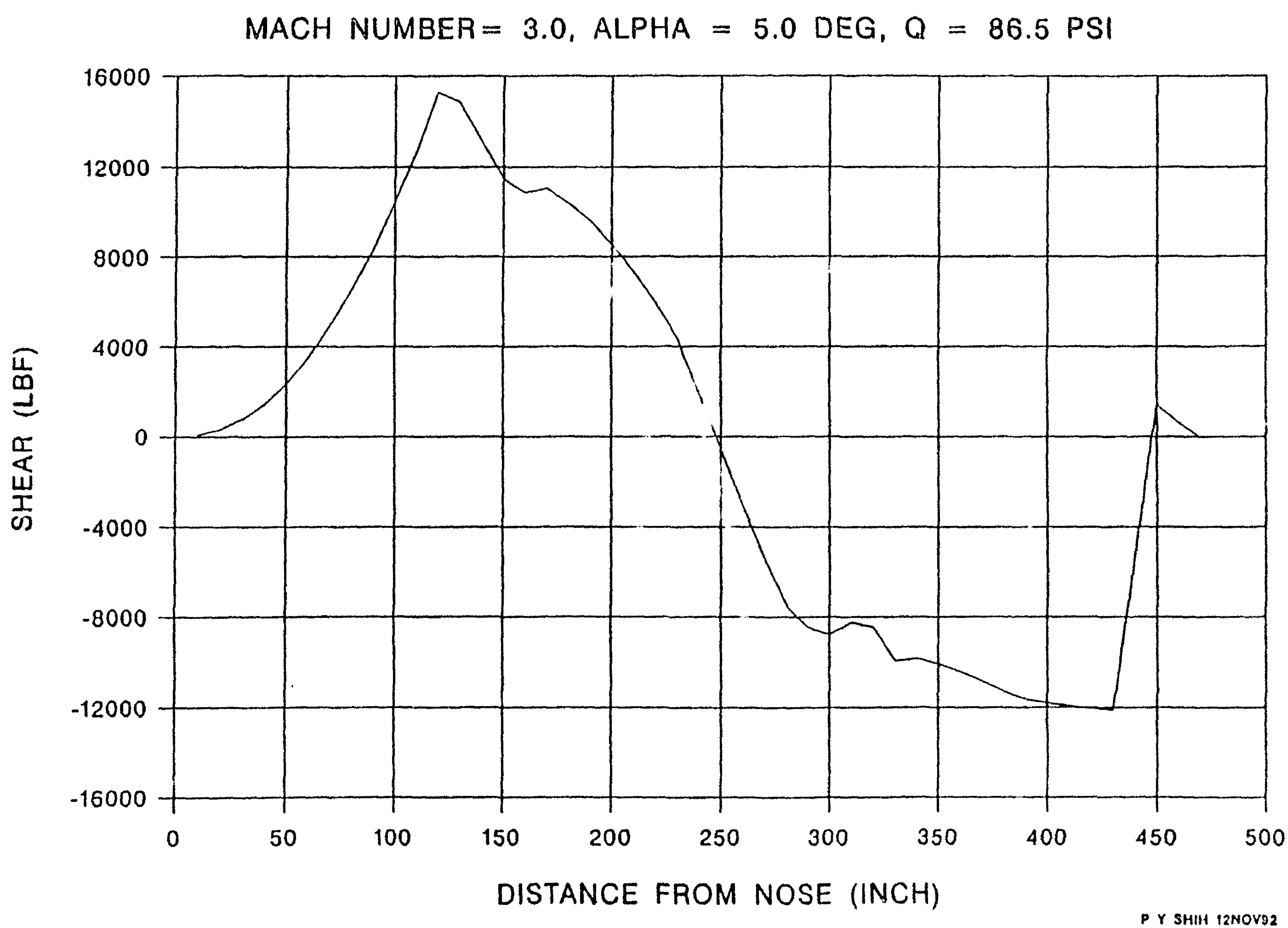

Figure 2.?-4 Shear of CLB Missile

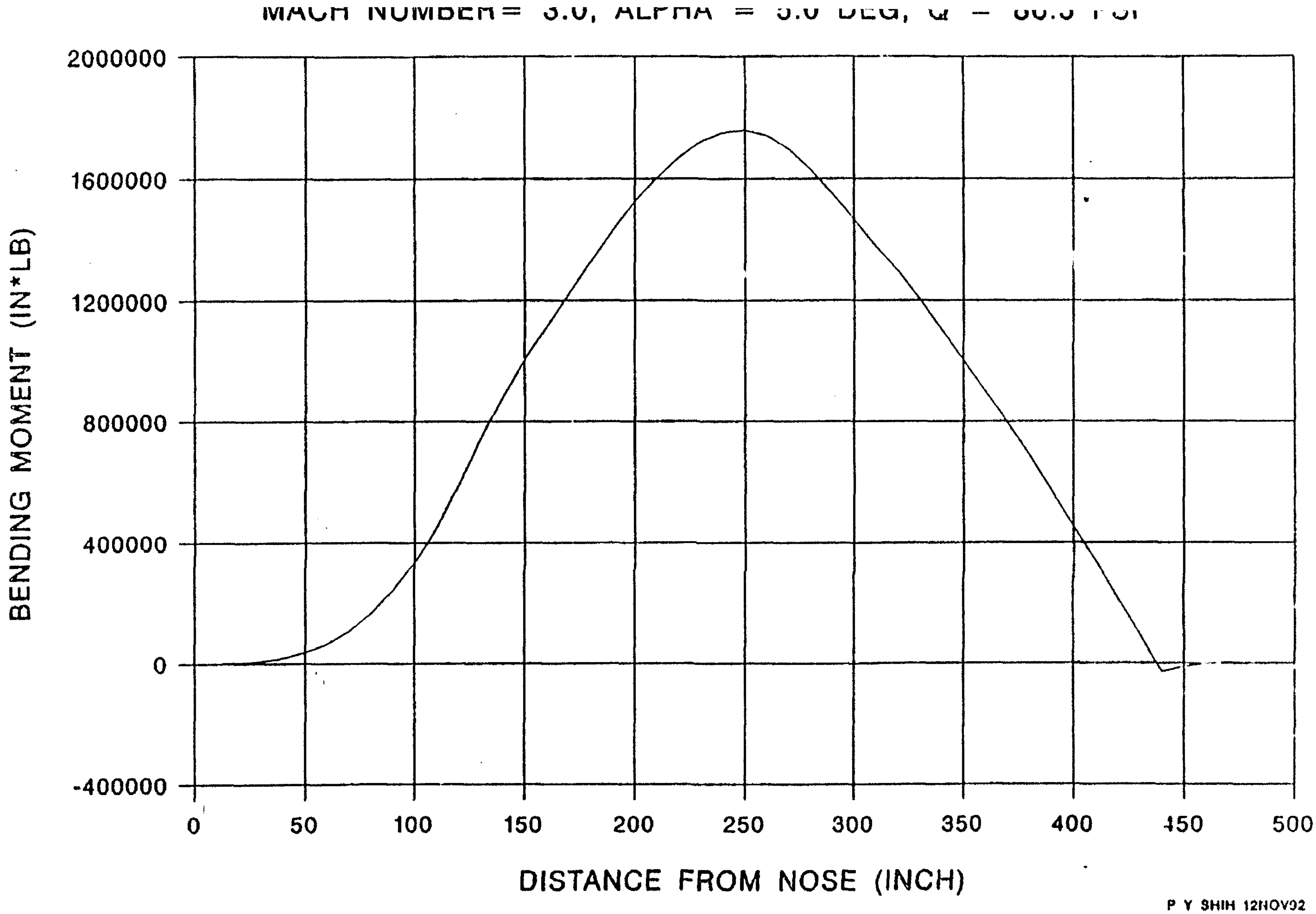

Figure 2.3-5 Bending Moment of CLB Missile

Figure 2.3-6 depicts a baseline configuration of 4 clustered missiles attached to a forebody (second stage) with a fin attachment ring at the aft end. This ring attaches to each motor through the existing fin post attachment sockets. Thrust loads are applied to a redistribution bulkhead as loads normal to the bulk head at station 100. The net applied load at the bulk head (rocket thrust-drag-inertia), is in the order of 150000 pounds. This load is then resisted by a structural interstage which transmits the load into the second stage.

2.3.2.2 Fin Support

The CLB fin planform is shown along with required geometry. The maximum total fin load carried into the motor structure is 24000 pounds (12000 pounds per fin) at 5 degrees angle of attack. This load produces a fin bending moment of approximately 126,000 inch-pounds which is introduced into the existing LANCE pivot socket. This socket has a tested strength of 35,800 inch pounds, (margin of safety of minus 71 percent) thus requiring a much larger socket and post in the motor case. The maximum CLB fin load that can be accommodated using the geometry shown is 2,,00 pounds per surface. Based on the above discussion, it becomes clear that a different structure for the fin is needed. The redistribution of the total fin load (24000 pounds) directly into the motor case structure appears highly doubtful. An arrangement as shown in the baseline cannot provide proper structural stiffness to meet fin flutter requirements. In view of these requirements it is necessary to have a more elaborate structure that will permit the vehicle to perform free of structural limitations imposed by the motor structure.

2.3.3 Modified Baseline Structural Arrangement

The previous discussion highlights some of the readily apparent structural problems associated with the baseline CLB configuration. This concept appears to be similar to one suggested by MICOM. Although these points may appear to condemn the CLB concept, which is not what is intended, they do point to a totally different structural concept which is designed to accept any measure of applied structural loading without limiting the vehicle design performance. This can be achieved by the following concept.

Figure 2.3-7 presents a structural modification of the baseline expressly conceived to overcome all the above limitations. It accomplishes this in the following manner:

A. Employs LANCE motors as thrusters only. Each motor case will transmit thrust, internal pressure, motor weight induced inertial loads, and TVC forces.

CLB – STRUCTURE

- BASELINE

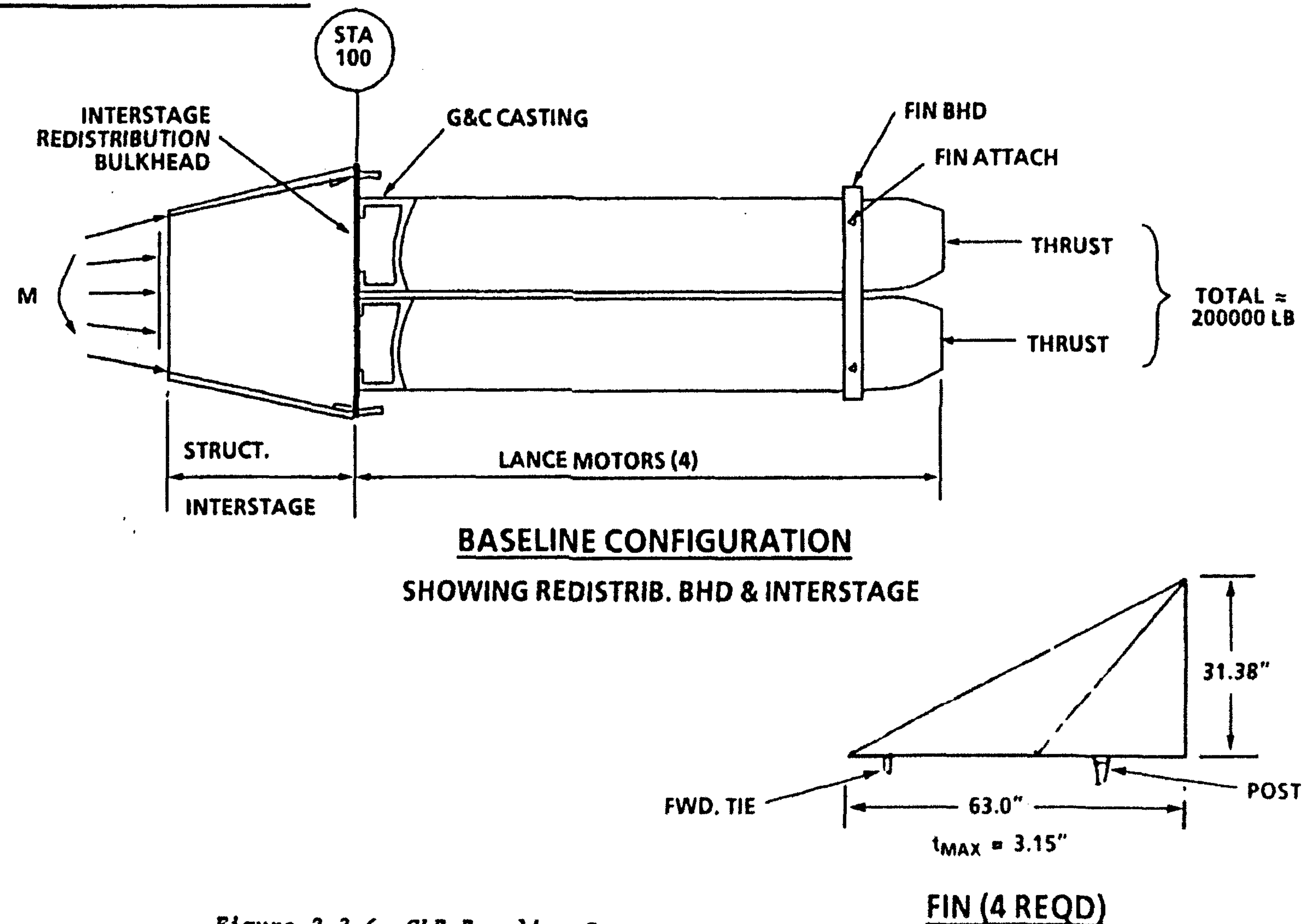

Figure 2.3-6 CLB Baseline Structure

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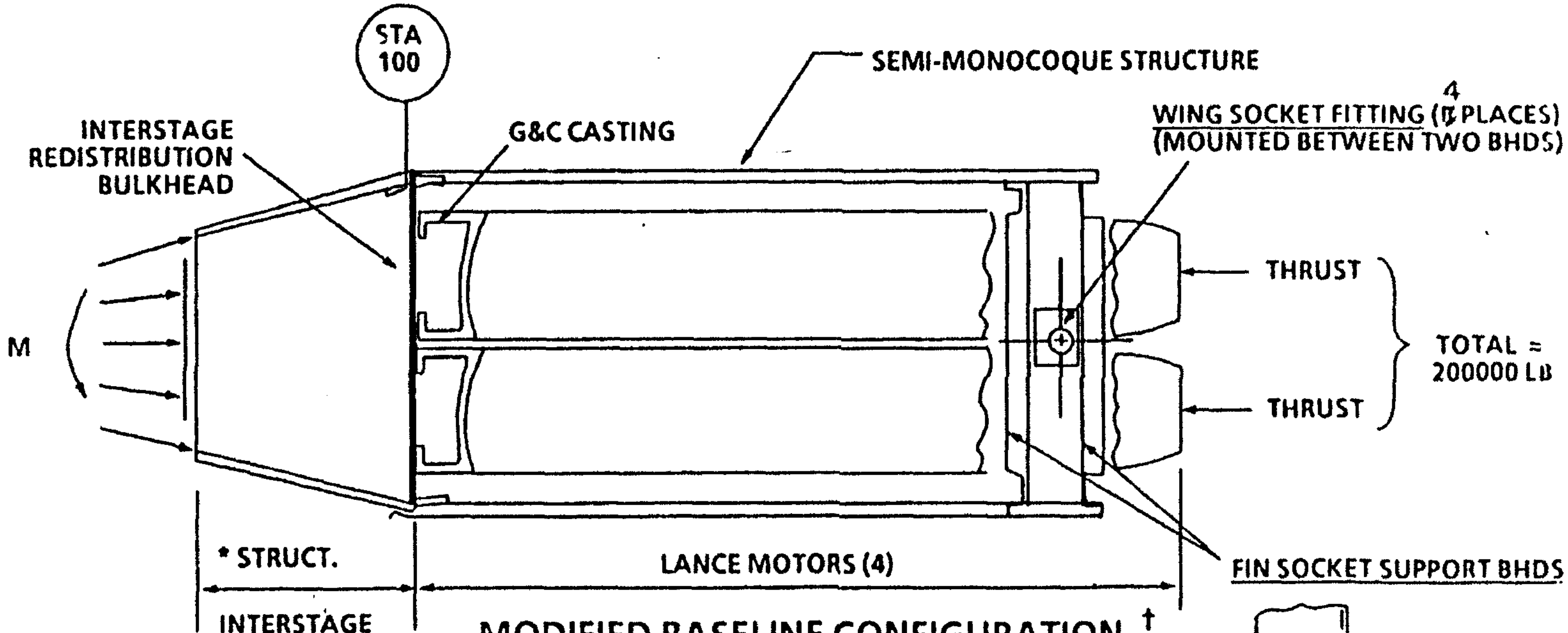
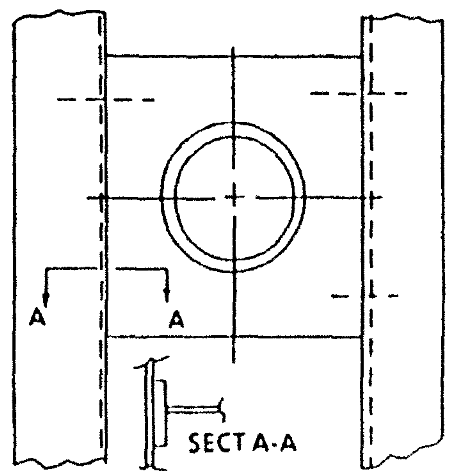
CLB – STRUCTURE

• MODIFIED BASELINE



MODIFIED BASELINE CONFIGURATION †

SHOWING REDISTRIB. BHD & INTERSTAGE



NOTE:
* STRUCT. INTERSAGE REDISTRIBUTES SECOND STAGE LOADS
† NOT SHOWN IS INTERNAL STRUCTURE FOR LAUNCH HOLDBACK AND LAUNCHER SUPPORT LOADS

Figure 2.3-7 CLB Modified Baseline Structure

FIN SOCKET FTG. ATT/[redacted]WIT[redacted]8" [redacted]TS

B. Provides necessary airframe structure to properly redistribute high concentrated loads from fins, launch holdback, control forces from the TVC system which are transmitted from the motors into the airframe and thrust loads.

C. Provides proper structural depth for overall vehicle stiffness to minimize aeroelastic problems and for fuselage stiffness at fin attachment points.

The structure envisioned is a semi-monocoque structure using bulkheads and frames for load redistribution. The fins will be a hybrid configuration with sufficient maximum depth to provide sufficient stiffness to produce a flutter free structure. Conventional airframe materials and joining will be used for the major portion of the vehicle.

2.4 Mass Properties

The estimated mass properties for the CLB/LFT-40 and CLB/LFT-22 clustered booster configurations are presented in the following tables. The estimates are based on available Lance data and design layouts.

Table 2.4-1 presents the weight and balance summary of both configurations giving details of the added components required to attach the boosters. The fin weight shown assumes the fins to be similar to the "large Lance fins" but approximately seventy percent larger in area.

Tables 2.4-2 and 2.4-3 present the mass properties of the vehicles during the booster burn phase.

Table 2.4-4 documents mass property data used for each Lance booster.

Table 2.4-5 provides mass distribution data for the CLB/LFT-40 configuration at the "Max Q" condition prepared for load estimation purposes.

Table 2.4-1 Clustered Booster Weight & Balance

Configuration / Item	CLB/LFT-40 Weight Lb	CLB/LFT-40 X Cg Ins	CLB/LFT-40 Moment Lb ins	CLB/LFT-22 Weight Lb	CLB/LFT-22 X Cg Ins	CLB/LFT-22 Moment Lb ins
Liquid Fueled Target	3988.0	207.0	825516	2710.0	158.5	429535
Booster Attachment						
Booster Attach Flange	90.7	320.2	29042	90.7	260.7	23645
Booster Attach Plate	126.8	323.1	40969	126.8	263.6	33424
Booster Attach Cone				177.0	247.0	43719
Fairing	115.5	306.8	35430	111.0	240.1	26651
Booster to Fairing Straps	54.3	323.6	17571	54.3	263.6	14313
Fairing to U/Stage Strap	8.4	279.2	2345	8.4	205.6	1727
Ring Grooved Clamp	302.5	318.5	96346	100.8	259.6	26168
Fairing linear Shaped Charge Syst	17.5	306.8	5368	17.5	233.8	4092
Attach Cone Pyro System				22.0	257.9	5674
Booster to Booster Aft Tube	6.3	448.5	2825	6.3	388.3	2446
Booster to Booster Aft Clamps	11.5	448.5	5157	11.5	388.3	4465
Misc 5 %	36.7	320.0	11736	36.3	260.0	9442
Booster (4)	9012.0	404.1	3641299	9012.0	344.2	3101660
Booster Fins (4)	120.0	441.0	52920	120.0	381.2	45744
Total Launch	13890.2	343.2	4766525	12604.6	299.3	3772706

Note: X Cg: Inches from theoretical Nose of LFT.

Table 2.4-2 Weight and Balance Summary.
Clustered Lance Booster CLB/LFT-40

Condition.	Time after Launch (Secs)	Weight (Lb)	Center of Gravity (Inches)			Moment of Inertia (Slugs Feet Squared)		
			X Cg	Y Cg	Z Cg	I Roll	I Pitch	I Yaw
Launch with 4 Boosters	0.00	13890.	343.2	0.0	0.0	776.2	29777.	29777.
Booster weight 2200 lb x 4	0.29	13678.	342.5	0.0	0.0	764.7	29637.	29637.
Booster weight 2000 lb x 4	1.26	12878.	339.8	0.0	0.0	721.0	29173.	29173.
Booster weight 1800 lb x 4	2.22	12078.	336.5	0.0	0.0	677.2	28637.	28637.
Booster weight 1600 lb x 4	3.22	11278.	332.3	0.0	0.0	633.5	27905.	27905.
Booster weight 1400 lb x 4	4.21	10478.	326.9	0.0	0.0	589.8	26832.	26832.
Booster weight 1200 lb x 4	5.19	9678.	319.8	0.0	0.0	546.1	25325.	25325.
Booster weight 1000 lb x 4	6.23	8878.	310.9	0.0	0.0	502.4	23328.	23328.
Booster weight 800 lb x 4	7.46	8078.	299.8	0.0	0.0	458.7	20750.	20750.
Booster weight 756 lb x 4 (Burnout)	8.00	7902.	297.1	0.0	0.0	449.1	20113.	20113.
Liquid Fueled Target (No Booster)		3988.	207.0	0.0	0.0	116.0	3418.	3418.

Note;
1. Center of gravity.
   X Cg: Inches from theoretical Nose of LFT.
   y Cg. Inches from Missile Centerline.
   z Cg. Inches from Missile Centerline.

2. Above Data include fins estimated at 30 lb each.

Table 2.4-3 Weight and Balance Summary.
Clustered Lance Booster CLB/LFT-22

Condition.	Time after Launch (Secs)	Weight (Lb)	Center of Gravity (Inches)			Moment of Inertia (Slugs Feet Squared)		
			X Cg	Y Cg	Z Cg	I Roll	I Pitch	I Yaw
Launch with 4 Boosters	0.00	12605.	299.3	0.0	0.0	675.5	20801.	20801.
Booster weight 2200 lb x 4	0.29	12393.	298.8	0.0	0.0	664.0	20715.	20715.
Booster weight 2000 lb x 4	1.26	11593.	297.0	0.0	0.0	620.2	20458.	20458.
Booster weight 1800 lb x 4	2.22	10793.	294.6	0.0	0.0	576.5	20177.	20177.
Booster weight 1600 lb x 4	3.22	9993.	291.3	0.0	0.0	532.8	19774.	19774.
Booster weight 1400 lb x 4	4.21	9193.	286.7	0.0	0.0	489.1	19134.	19134.
Booster weight 1200 lb x 4	5.19	8393.	280.4	0.0	0.0	445.4	18175.	18175.
Booster weight 1000 lb x 4	6.23	7593.	272.2	0.0	0.0	401.7	16843.	16843.
Booster weight 800 lb x 4	7.46	6793.	261.5	0.0	0.0	358.0	15052.	15052.
Booster weight 756 lb x 4 (Burnout)	8.00	6617.	258.8	0.0	0.0	348.4	14601.	14601.
Liquid Fueled Target (No Booster)		2710.	158.5	0.0	0.0	15.0	1943.	1943.

Note:
1. Center of gravity.
X Cg: Inches from theoretical Nose of LFT.
y Cg. Inches from Missile Centerline.
z Cg. Inches from Missile Centerline.

Table 2.4-4 BOOSTER MASS PROPERTY DATA.
LANCE PROPULSION SYSTEM (MODIFIED)

Weight Lb	Xcg MSL (1)	Moment of Inertia (Slugs ft^2) I Roll	I Pitch	I Yaw
2253	180.6	11.3	614	614
2200	181.0	11.3	600	600
2000	182.9	11.3	557	557
1800	184.8	11.3	527	527
1600	186.4	11.3	502	502
1400	187.2	11.3	478	478
1200	186.9	11.3	449	449
1000	185.3	11.3	409	409
800	181.8	11.3	353	353
756	180.8	11.3	338	338

NOTE: X cg is shown in the original LANCE Reference System.
(Forward end of Propulsion System LANCE Station MS 100)

TABLE 2.4-5 Mass Distribution Data Configuration CLB/LFT-40
At Time = 7.05 Seconds, Estimated Max Q Condition

UNITS ARE WEIGHT - POUNDS
CENTER OF GRAVITY - INCHES
MOMENTS AND PRODUCTS OF INERTIA - POUND-INCHES SQUARED

BAY	WEIGHT		C.G.	MOMENTS OF INERTIA		PRODUCTS OF INERTIA	
0.0	56.321	X	26.53	IXX	0.6296667E+04	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.1628328E+05	PXY	0.0000000E+00
50.0	1.126	Z	0.00	IZZ	0.1628328E+05	PYZ	0.0000000E+00
50.0	44.152	X	64.81	IXX	0.5341226E+04	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.5166002E+04	PXY	0.0000000E+00
75.0	1.766	Z	0.00	IZZ	0.5166002E+04	PYZ	0.0000000E+00
75.0	79.095	X	87.58	IXX	0.7537596E+04	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.7880411E+04	PXY	0.0000000E+00
100.0	3.164	Z	0.00	IZZ	0.7880411E+04	PYZ	0.0000000E+00
100.0	81.700	X	112.52	IXX	0.1049754E+05	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.9298388E+04	PXY	0.0000000E+00
123.0	3.552	Z	0.00	IZ[illegible]	0.9298388E+04	PYZ	0.0000000E+00
123.0	773.244	X	138.91	I[illegible]	0.1296715E+06	PXZ	0.0000000E+00
TO		Y	0.00	I[illegible]	0.1309586E+06	PXY	0.0000000E+00
155.5	23.792	Z	0.00	IZZ	0.1309586E+06	PYZ	0.0000000E+00
155.5	82.771	X	161.35	IXX	0.214[illegible]205E+05	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.1230956E+05	PXY	0.0000000E+00
167.0	7.197	Z	0.00	IZZ	0.1230956E+05	PYZ	0.0000000E+00
167.0	390.672	X	180 4[illegible]	IXX	0.4693557E+05	PXZ	0.0000000E+00
TO		Y	[illegible]	IYY	0.4212623E+05	PXY	0.0000000E+00
192.0	15.627	Z	[illegible] 00	IZZ	0.4212623E+05	PYZ	0.0000000E+00
192.0	402.867	X	[illegible]03.57	IXX	0.4508738E+05	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.3997768E+05	PXY	0.0000000E+00
214.0	13.312	Z	0.00	IZZ	0.3997768E+05	PYZ	0.0000000E+00
214.0	298.[illegible]9	X	222.93	IXX	0.3082040E+05	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.2085264E+05	PXY	0.0000000E+00
229.0	1[illegible].877	Z	0.00	IZZ	0.2085264E+05	PYZ	0.0000000E+00
229.0	699.384	X	241.50	IXX	0.4437095E+05	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.5863167E+05	PXY	0.0000000E+00
254.0	27.975	Z	0.00	IZZ	0.5863167E+05	PYZ	0.0000000E+00
254.0	640.358	X	265.47	IXX	0.4311580E+05	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.4939798E+05	PXY	0.0000000E+00
277.0	27.842	Z	0.00	IZZ	0.4939798E+05	PYZ	0.00[illegible]000E+00
277.0	434.812	X	287.54	IXX	0.1159345E+06	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.8025833E+05	PXY	0.0000000E+00
302.0	17.392	Z	0.00	IZZ	0.8025833E+05	PYZ	0.0000000E+00
302.0	236.765	X	313.10	IXX	0.7641193E+05	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.4299217E+05	PXY	0.0000000E+00
317.7	15.081	Z	0.00	IZZ	0.4299217E+05	PYZ	0.0000000E+00
317.7	469.393	X	320.89	IXX	0.1555063E+06	PXZ	0.0000000E+00
TO		Y	0.00	IYY	0.7921416E+05	PXY	0.0000000E+00
323.5	80.930	Z	0.00	IZZ	0.7921416E+05	PYZ	0.0000000E+00

TABLE 2.4-5 (Continued) Mass Distribution Data Configuration CLB/LFT-40
At Time = 7.05 Seconds, Estimated Max Q Condition

UNITS ARE WEIGHT - POUNDS
CENTER OF GRAVITY - INCHES
MOMENTS AND PRODUCTS OF INERTIA - POUND-INCHES SQUARED

BAY	WEIGHT		C.G.	MOMENTS OF INERTIA		PRODUCTS OF INERTIA	
323.5	356.919	X	333.03	IXX	0.1307140E+06	PXZ	0.3427707E-26
TO		Y	0.00	IYY	0.7198715E+05	PXY	0.4561818E-26
340.0	21.631	Z	0.00	IZZ	0.7369156E+05	PYZ	-0.5361789E-30
340.0	606.619	X	351.45	IXX	0.1881056E+06	PXZ	-0.1220553E+02
TO		Y	0.00	IYY	0.1279897E+06	PXY	0.1890931E-25
365.0	24.265	Z	-0.01	IZZ	0.1313640E+06	PYZ	-0.2781742E-27
365.0	485.714	X	375.88	IXX	0.1498160E+06	PXZ	0.3785507E+01
TO		Y	0.00	IYY	0.9229624E+05	PXY	0.2072235E-25
386.0	23.129	Z	-0.01	IZZ	0.9499808E+05	PYZ	-0.2474960E-27
386.0	481.464	X	398.02	IXX	0.1458558E+06	PXZ	-0.1703201E-25
TO		Y	0.00	IYY	0.9828272E+05	PXY	0.1337093E-26
411.0	19.259	Z	0.00	IZZ	0.1009609E+06	PYZ	0.7888609E-30
411.0	460.252	X	423.50	IXX	0.1332331E+06	PXZ	-0.1061838E-25
TO		Y	0.00	IYY	0.8924635E+05	PXY	0.1939536E-25
436.0	18.410	Z	0.00	IZZ	0.9180651E+05	PYZ	0.0000000E+00
436.0	416.276	X	442.80	IXX	0.3257664E+06	PXZ	0.1643040E-26
TO		Y	0.00	IYY	0.1683688E+06	PXY	-0.9996286E-27
449.0	32.021	Z	0.00	IZZ	0.1699574E+06	PYZ	-0.1577722E-29
449.0	866.064	X	459.99	IXX	0.3166314E+06	PXZ	0.4710671E-25
TO		Y	0.00	IYY	0.1930547E+06	PXY	-0.1756847E-25
468.8	43.741	Z	0.00	IZZ	0.1978326E+06	PYZ	0.1269573E-29

EIGHT	C.G.		MOMENTS OF INERTIA		PRODUCTS OF INERTIA	
	X	302.855	IXX	0.2129072E+07	PXZ	-0.6692935E+03
8363.000	Y	0.000	IYY	0.9778673E+08	PXY	0.4010876E-25
	Z	-0.001	IZZ	0.9780611E+08	PYZ	-0.5248156E-27

2.5 Propulsion System Performance

2.5.1 Propulsion System Forces and Moments

Differences in thrust magnitude and alignment between the four propulsion systems in the cluster will cause side forces and pitch, yaw and roll moments on the vehicle. The coordinate system used to evaluate these forces and moments is shown in figure 2.5-1.

Two cases were evaluated. In one case the four propulsion system tank pressures were independent of each other. Since each propulsion system has its own Solid Propellant Gas Generator (SPGG) and Hot Gas Relief Valve (HGRV), the propulsion systems can have significant differences in tank pressure (and thrust) at any given time. The results for this case are given in table 2.5-1.

In the second case, it was assumed that the four propulsion systems have the same SPGG pressure at any given time. This could be accomplished by means of a manifold which would allow the flow of SPGG gases between the propulsion systems. The results for this case are given in table 2.5-2.

2.5.1.1 Ignition Transient

Time in tables 2.5-1 and 2.5-2 is measured from ignition of the last of the four engines in the cluster. Even though the ignition signal will be applied simultaneously to the four propulsion systems, the engines will not necessarily ignite at the same time. Ignition advances through a series of steps, culminating in ignition of the SPGG which pressurizes the feed system. As the feed system pressure increases, the fuel diaphragm ruptures at about 600 psig, and the oxidizer diaphragm ruptures at about 700 psig, allowing propellants to flow into the engine where they ignite and provide thrust. The time between the ignition signal and engine ignition has an average value of 0.291 seconds with a standard deviation of 0.023 seconds when the propulsion system is at ambient temperature (59 degrees F ± 15 degrees F). An illustration of thrust variation during the ignition transient is given in figure 2.5-2.

The forces and moments at ignition of the fourth engine were determined by Monte Carlo techniques. Data used in the computations are given in table 2.5-3. The results of the computations are given in table 2.5-1 and 2.5-2 at time = 0.

2.5.1.2 Lance Axial Thrust

Axial thrust after engine ignition was derived from SPGG pressure data from static and flight tests. The average SPGG pressure is given in figure 2.5-3 for propulsion system temperatures of -40, 59 and 140 degrees F. The standard deviation of SPGG pressure is given in figure 2.5-4. The shape of the SPGG pressure curve was modified to account for

Table 2.5-1

Forces and Moments When Tank Pressures Are Independent of Each Other

T	FXLN	FXLNSD	MALSD	L	LSD	FXCL	FXCLSD	FSSD	MXSD	MPYSD
0.000			1.40	11.25	0.05	127113	6709	89	118	6128
0.400	48564	2440	1.40	11.25	0.05	194256	4879	136	182	4592
0.600	50472	1600	1.41	11.25	0.05	201888	3200	142	190	3029
0.800	51059	1287	1.41	11.25	0.05	204236	2575	144	192	2451
1.000	51380	1085	1.42	11.25	0.05	205520	2170	146	195	2079
1.200	51349	957	1.42	11.25	0.05	205396	1915	146	195	1845
1.350	51367	915	1.43	11.25	0.05	205468	1829	147	196	1767
1.500	51390	963	1.43	11.25	0.05	205560	1926	147	196	1856
1.600	51406	1076	1.43	11.25	0.05	205624	2152	147	196	2062
1.800	50828	1341	1.44	11.25	0.05	203312	2681	146	196	2549
2.000	49995	896	1.45	11.25	0.05	199980	1792	145	194	1731
2.200	49139	673	1.45	11.25	0.05	196556	1345	143	190	1326
2.400	48944	601	1.46	11.25	0.05	195776	1202	143	191	1198
2.650	48660	559	1.47	11.25	0.05	194640	1118	143	191	1123
3.000	48645	589	1.49	11.25	0.05	194580	1177	145	194	1176
3.500	49000	633	1.52	11.25	0.05	196000	1266	149	199	1256
4.000	49246	696	1.56	11.25	0.05	196984	1391	154	205	1367
4.500	49529	770	1.61	11.25	0.05	198116	1540	159	213	1502
5.000	49792	908	1.66	11.25	0.05	199168	1815	165	221	1752
5.325	49750	1035	1.69	11.25	0.05	199000	2071	168	225	1985
6.000	45156	962	1.76	11.25	0.05	180624	1925	159	212	1843
6.500	41538	966	1.82	11.25	0.05	166152	1931	151	202	1843
7.000	39341	937	1.88	11.25	0.05	157364	1873	148	198	1787
8.000	34957	941	2.00	11.25	0.05	139828	1882	140	187	1788

DEFINITION OF COLUMN HEADINGS

T - TIME FROM IGNITION OF FOURTH ENGINE, SEC
FXLN - NOMIAL AXIAL THRUST OF ONE LANCE, LB
FXLNSD - STD DEV OF FXLN, LB
MALSD - STD DEV OF THRUST MALALIGNMENT, MILLIRADIANS
L - NOMINAL DISTANCE FROM LANCE CENTERLINE TO PITCH AND YAW AXES, IN.
LSD - STD DEV OF L, IN.
FXCL - NOMINAL AXIAL THRUST OF THE CLUSTER, LB
FXCLSD - STD DEV OF FXCL, LB
FSSD - STD DEV OF CLUSTER SIDE FORCE, LB
MXSD - STD DEV OF CLUSTER ROLL MOMENT, LB-FT
MPYSD - STD DEV OF CLUSTER PITCH AND YAW MOMENTS, LB-FT

Table 2.5-2

Forces and Moments When the Four Propulsion Systems Have the Same Pressures

T	FXLN	FXLNSD	MALSD	L	LSD	FXCL	FXCLSD	FSSD	MXSD	MPYSD
0.000	28704	350	1.40	11.25	0.05	114816	700	80	107	698
0.400	48564	350	1.40	11.25	0.05	194256	700	136	182	771
0.600	50472	350	1.41	11.25	0.05	201888	700	142	190	779
0.800	51059	350	1.41	11.25	0.05	204236	700	144	192	782
1.000	51380	350	1.42	11.25	0.05	205520	700	146	195	784
1.200	51349	350	1.42	11.25	0.05	205396	700	146	195	783
1.350	51367	350	1.43	11.25	0.05	205468	700	147	196	784
1.500	51390	350	1.43	11.25	0.05	205560	700	147	196	784
1.600	51406	350	1.43	11.25	0.05	205624	700	147	196	784
1.800	50828	350	1.44	11.25	0.05	203312	700	146	196	781
2.000	49995	350	1.45	11.25	0.05	199980	700	145	194	777
2.200	49139	350	1.45	11.25	0.05	196556	700	143	190	774
2.400	48944	350	1.46	11.25	0.05	195776	700	143	191	773
2.650	48660	350	1.47	11.25	0.05	194640	700	143	191	771
3.000	48645	350	1.49	11.25	0.05	194580	700	145	194	771
3.500	49000	350	1.52	11.25	0.05	196000	700	149	199	773
4.000	49246	350	1.56	11.25	0.05	196984	700	154	205	774
4.500	49529	350	1.61	11.25	0.05	198116	700	159	213	775
5.000	49792	350	1.66	11.25	0.05	199168	700	165	221	776
5.325	49750	350	1.69	11.25	0.05	199000	700	168	225	776
6.000	45156	350	1.76	11.25	0.05	180624	700	159	212	756
6.500	41538	350	1.82	11.25	0.05	166152	700	151	202	742
7.000	39341	350	1.88	11.25	0.05	157364	700	148	198	734
8.000	34957	350	2.00	11.25	0.05	139828	700	140	187	718

DEFINITION OF COLUMN HEADINGS

T - TIME FROM IGNITION OF FOURTH ENGINE, SEC
FXLN - NOMIAL AXIAL THRUST OF ONE LANCE, LB
FXLNSD - STD DEV OF FXLN, LB
MALSD - STD DEV OF THRUST MALALIGNMENT, MILLIRADIANS
L - NOMINAL DISTANCE FROM LANCE CENTERLINE TO PITCH AND YAW AXES, IN.
LSD - STD DEV OF L, IN.
FXCL - NOMINAL AXIAL THRUST OF THE CLUSTER, LB
FXCLSD - STD DEV OF FXCL, LB
FSSD - STD DEV OF CLUSTER SIDE FORCE, LB
MXSD - STD DEV OF CLUSTER ROLL MOMENT, LB-FT
MPYSD - STD DEV OF CLUSTER PITCH AND YAW MOMENTS, LB-FT

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ATTN: CSSD-IM-PA
Huntsville, AL 35807

1. Reference: DoD Directive 5230.24, Distribution Statements on Technical Documents, 18 Mar 87.

2. The Defense Technical Information Center received the enclosed report (referenced below) which is not marked in accordance with the above reference.

Clustered Lance Booster (CLB)
Contract no. DASG60-92-C-0120
27 Jan 93

3. We request the appropriate distribution statement be assigned and the report returned to DTIC within 5 working days.

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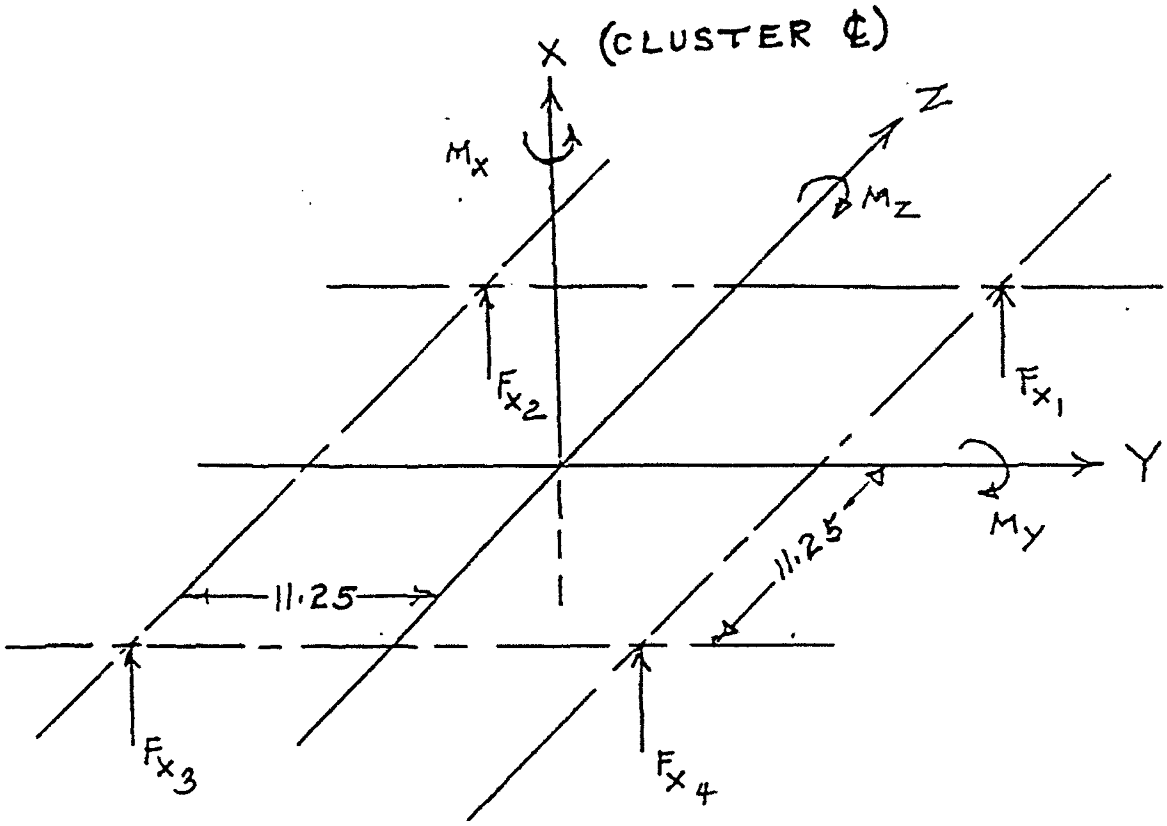
Table 2.5-3

Data Used to Compute Forces and Moments at Ignition of Fourth Engine

LINE NO.	PARAMETER	UNITS	MEAN	STD DEV	SOURCE OF DATA
1	Time from SPGG Ignition Signal to Engine Ignition	sec	0.291	0.023	Ambient Static Tests
2	Oxidizer Deadspace Pressure at Engine Ignition	psig	695	28.3	Oxidizer Burst Diaphragm Specification
3	Thrust/(Oxidizer Deadspace Press.) at Engine Ignition	lbs/psig	41.3	----	Lance Math Model
4	Variation in thrust for Constant Oxidizer Deadspace Pressure	lbs	----	350	Engine System Tests
5	Thrust at Engine Ignition	lbs	28704	1220	Computed from Lines 2, 3 and 4
6	Rate of Pressure Increase Immediately After Engine Ignition	psig/sec	3164	523.4	Ambient Static Tests
7	Rate of Thrust Increase Immediately After Engine Ignition	lbs/sec	130673	21616	Computed from Lines 3 and 6
8	Thrust Malalignment	milliradians	0.0	1.4	Engine System Tests
9	Thrust Vector Offset From Vehicle Y- and Z-Axis	inchs	0.05	0.05	Vehicle Design and Engine System Tests



Note: Origin of the coordinate system is located at Lance Missile Station 240.96, the point at which TVC side force is applied.

Figure 2.5-1 Coordinate System Used to Estimate Propulsion System Forces and Moments

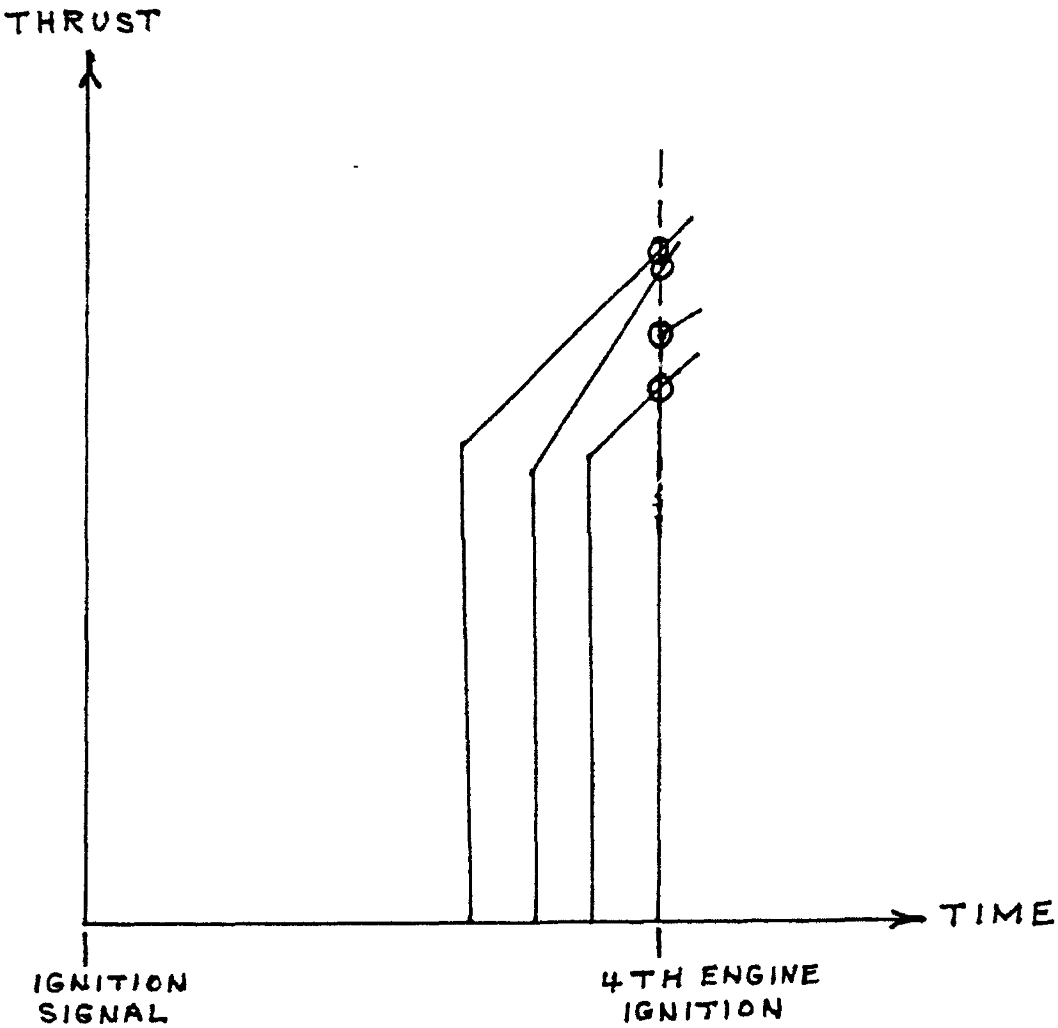


Figure 2.5-2 Illustration of Thrust Variations During Ignition Transient

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7½ X 10 INCHES
KEUFFEL & ESSER CO.

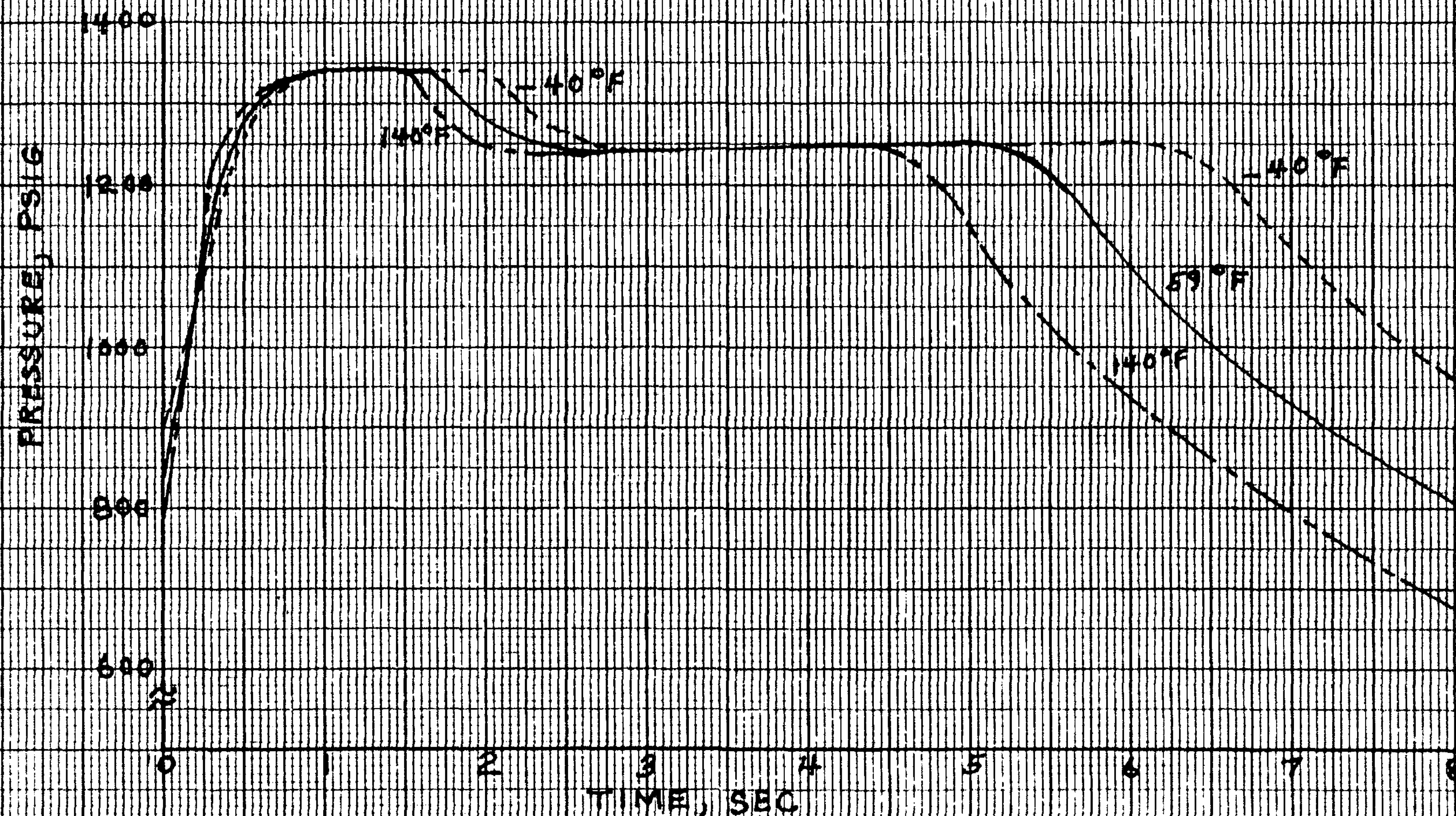

Figure 2.5-3 Estimated SPGG Pressure for System Temperatures of -40, 59 and 140 Degrees Fahrenheit

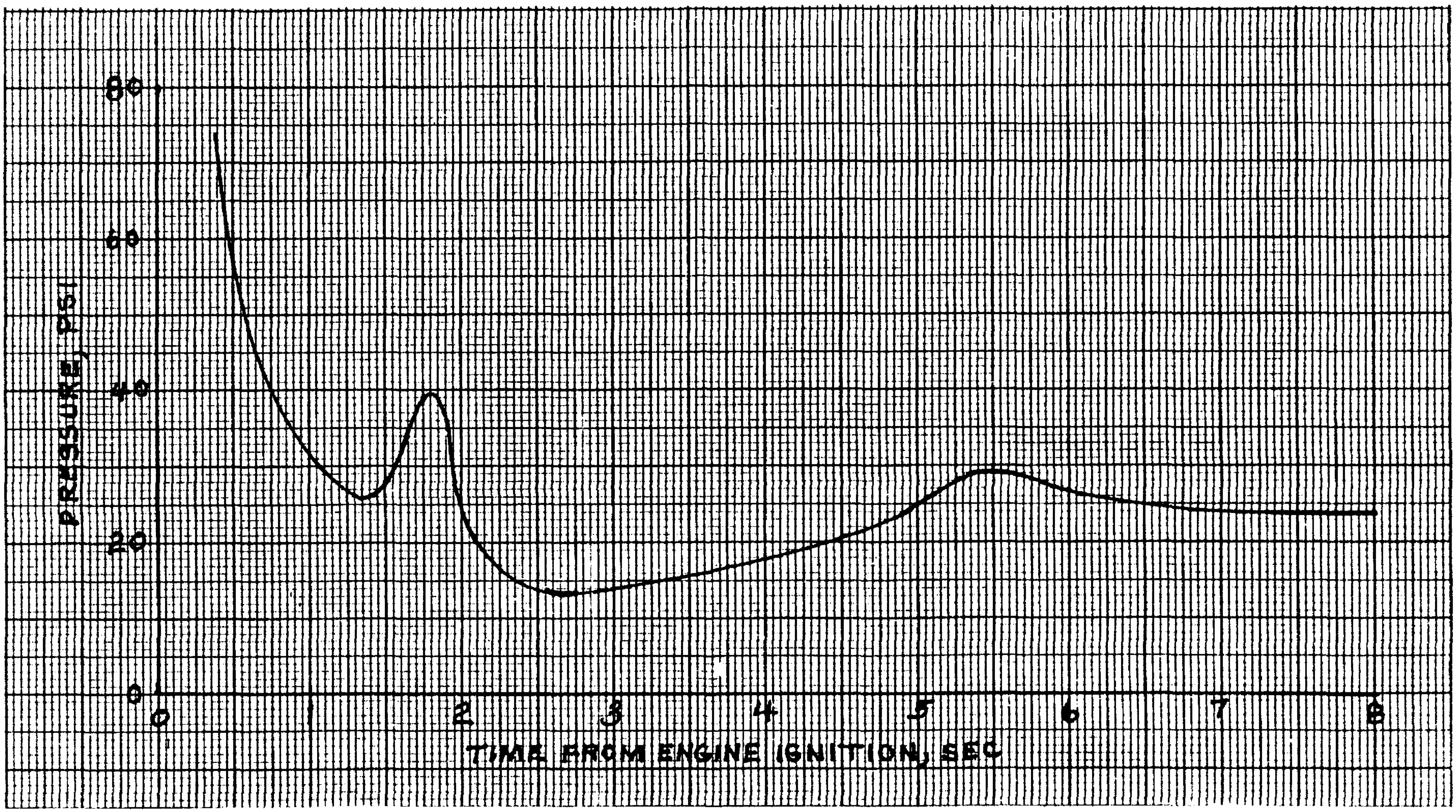

Figure 2.5-4 Standard Deviation of SPGG Pressure at Ambient Conditions

the absence of spin system flow on the CLB mission.

The SPGG pressure shown in figure 2.5-3 for a temperature of 59 degrees F was converted to the axial thrust shown in tables 2.5-1 and 2.5-2 by means of the Lance math model. The standard deviation of axial thrust in table 2.5-1 was computed by statistically combining the effects of SPGG pressure variation in figure 2.5-4 with the standard deviation of thrust when SPGG pressure is constant. The standard deviation of thrust when SPGG pressure is constant was determined to be 350 pounds from the analysis of engine test data. Since the data in table 2.5-2 is based on the four propulsion systems having the same SPGG pressure at any given time, the standard deviation of the axial thrust shown in table 2.5-2 is 350 pounds.

2.5.1.3 Lance Thrust Malalignment and Offset

The propulsion system centerline will be aligned as closely as practical with the x-axis shown in the figure 2.5-1 coordinate system. The malalignment of the thrust vector with respect to the x-axis will be the combination of thrust malalignment with respect to the propulsion system centerline and malalignment of the centerline with respect to the x-axis. The average thrust malalignment is assumed to be zero. The estimated standard deviation of thrust malalignment is given in tables 2.5-1 and 2.5-2.

Design considerations dictated that the propulsion system centerlines be located 11.25 inches from the Y- and Z-axes indicated in figure 2.5-1. The standard deviation of the thrust vector offset from this location was estimated to be 0.05 inch. Again, this standard deviation is a combination of mechanical tolerances on the location of the propulsion system centerline and offset of the thrust vector with respect to the propulsion system centerline.

2.5.1.4 Forces and Moments Acting on the Cluster

Conventional equations for computing forces and moments and root-sum-square techniques were used to convert the thrust variances of the individual propulsion systems into forces and moments acting on the cluster. Standard deviations of the forces and moments acting on the cluster are given in tables 2.5-1 and 2.5-2. Symmetry in the X-Y and X-Z planes was assumed in making these computations.

2.5.2 Thrust Vectoring

There are two methods available for vectoring the thrust of the CLB. One method is to use the Lance Thrust Vector Control (TVC) system which vectors the thrust by injecting fuel into the Lance booster nozzle through one of the four TVC valves on each engine. The other method is to control the thrust of each of the four throttlable sustainer engines in the cluster. A discussion of each of the methods follows.

2.5.2.1 The Lance TVC System

The maximum available TVC side force for the CLB shown in figure 2.5-5 was computed by assuming that TVC side force is proportional to axial thrust. Test data indicated that TVC side force was 390 pounds when thrust was 50,000 pounds. Thus, the CLB axial thrust in tables 2.5-1 and 2.5-2 was multiplied by 390/50,000 to obtain the maximum available TVC side force shown in figure 2.5-5.

Some of the fuel loaded into the Lance propulsion system was allocated for TVC use. After the allocated fuel has been used, each additional pound of fuel used by the TVC system reduces the utilization of oxidizer by about three pounds. This is illustrated in figure 2.5-6. After about 18 pounds has been used by one propulsion system (72 pounds by the CLB) available total impulse drops rapidly.

Specific impulse for the TVC system is a function of duty cycle and alignment of the required side force with the TVC ports. The fuel flow rate corresponding to 390 pounds of side force is 5.75 pounds per second, which yields a specific impulse of 67.8 seconds (390/5.75). This specific impulse can be attained only for a high duty cycle and if the required force is aligned with a TVC port. For example, if the required side force is located midway between two TVC ports (45 degrees from each port) the effective component would be 0.707 (sine or cosine of 45 degrees) of the TVC side force, which would reduce the specific impulse to 47.9 (0.707 x 67.8) seconds. With regards to duty cycle, TVC fuel usage becomes important only when the duty cycle is high, in which case the effective specific impulse approaches the ideal specific impulse. For the purposes of this study an effective specific impulse of 50 seconds will be used to determine TVC fuel usage in the sections which follow.

Estimated TVC usage is given in figures 2.5-7 and 2.5-8 for independent and equalized tank pressures, respectively. Figure 2.5-7 shows that there is a small probability (about 1 in 40) that TVC usage will exceed the 72 pound limit. This may be acceptable, depending upon the criticality of the total impulse requirement. Figure 2.5-8 indicates that TVC usage with equalized tank pressures is well within the 72 pound limit.

2.5.2.2 Sustainer Engine Thrust Control

The sustainer engine can be throttled between 0 and 100 percent thrust. This capability could be used to equalize the thrust of the four propulsion systems and to provide the pitch and yaw moments required to compensate for other disturbances.

The maximum thrust of one sustainer engine is given in figure 2.5-9 as a function of time.

When the fuel burst diaphragm ruptures, the fuel manifold pressure forces the sustainer engine pintle to the open position where it remains until the control cavities are primed. Priming is not completed until

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Figure 2.5-5 Maximum TVC Side Force as Function of Time

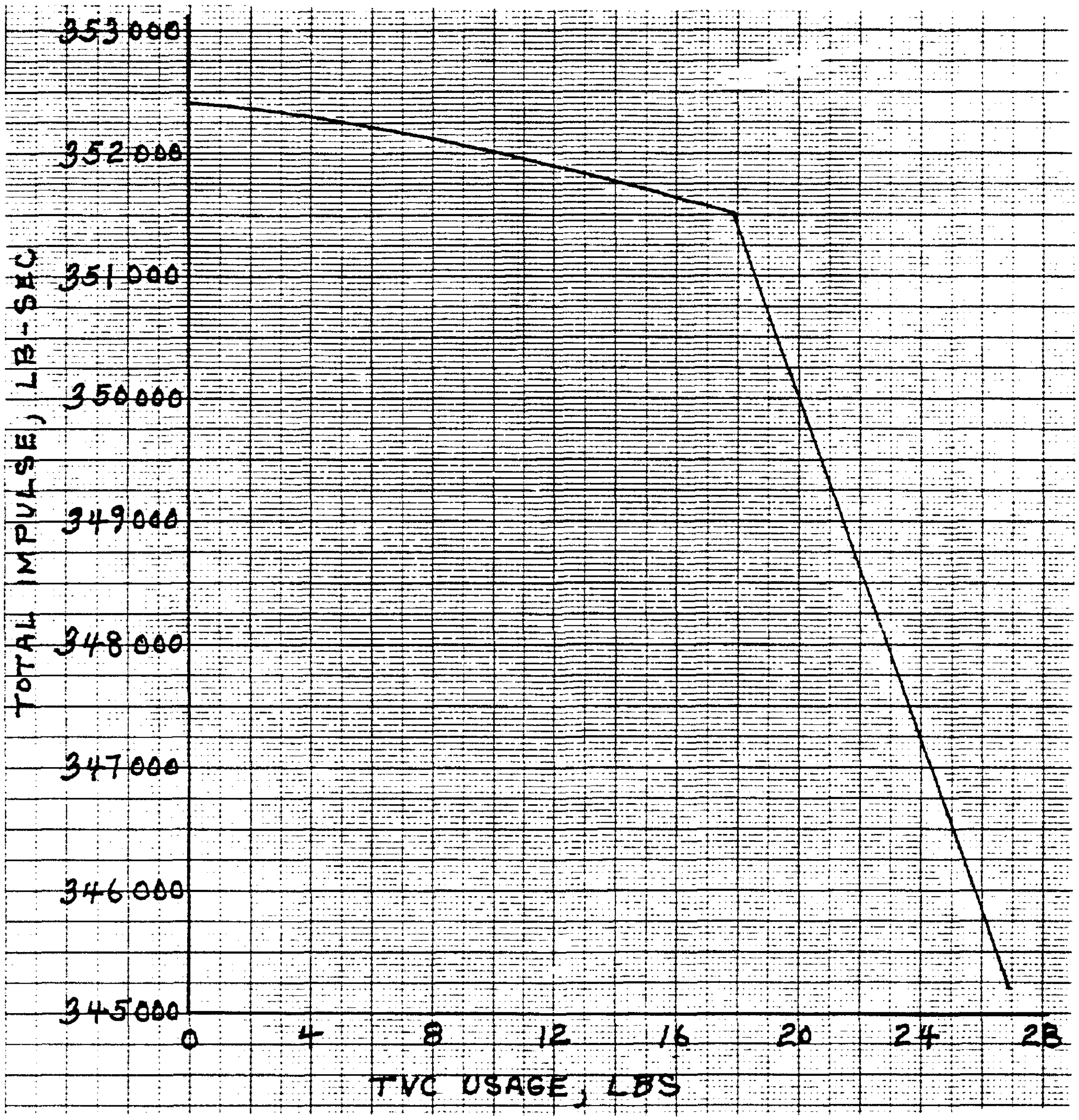

Figure 2.5-6 The Effects of TVC Fuel Usage on Total Impulse (One Lance)

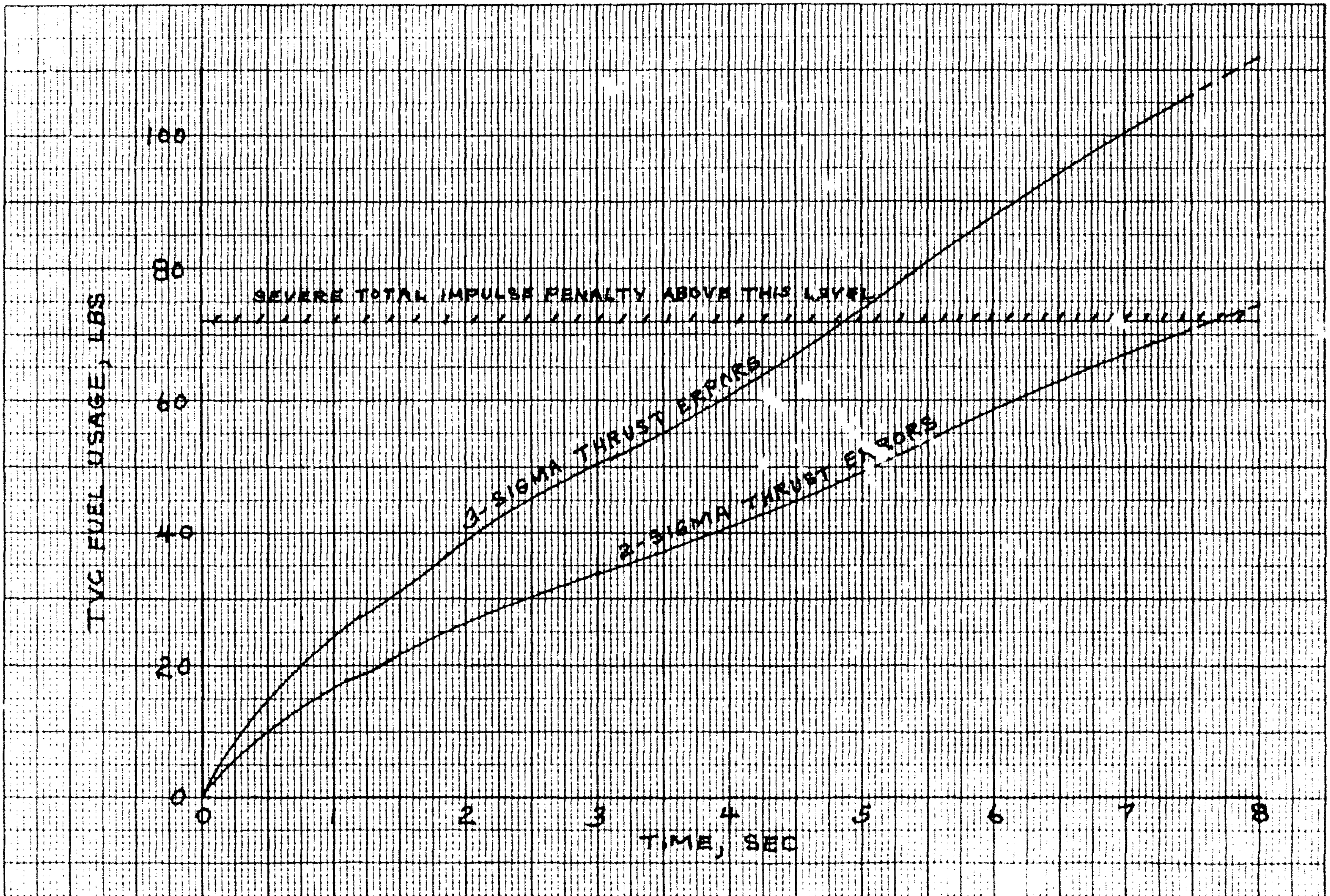

Figure 2.5-7 TVC Fuel Usage Required to Correct Thrust Errors When Tank Pressures Are Independent

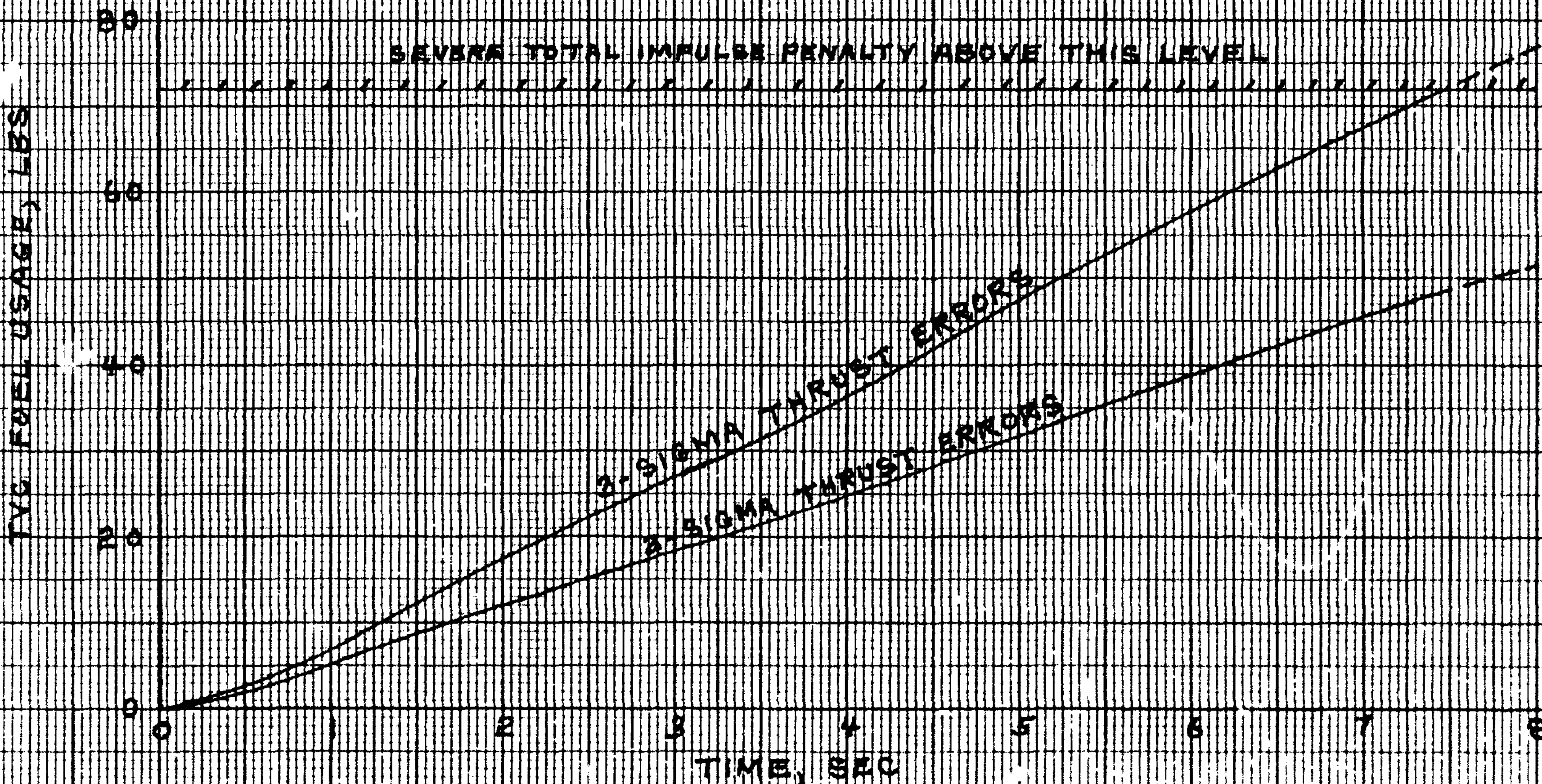

Figure 2.5-8 TVC Fuel Usage Required to Correct Thrust Errors When Tank Pressures Are Equalized

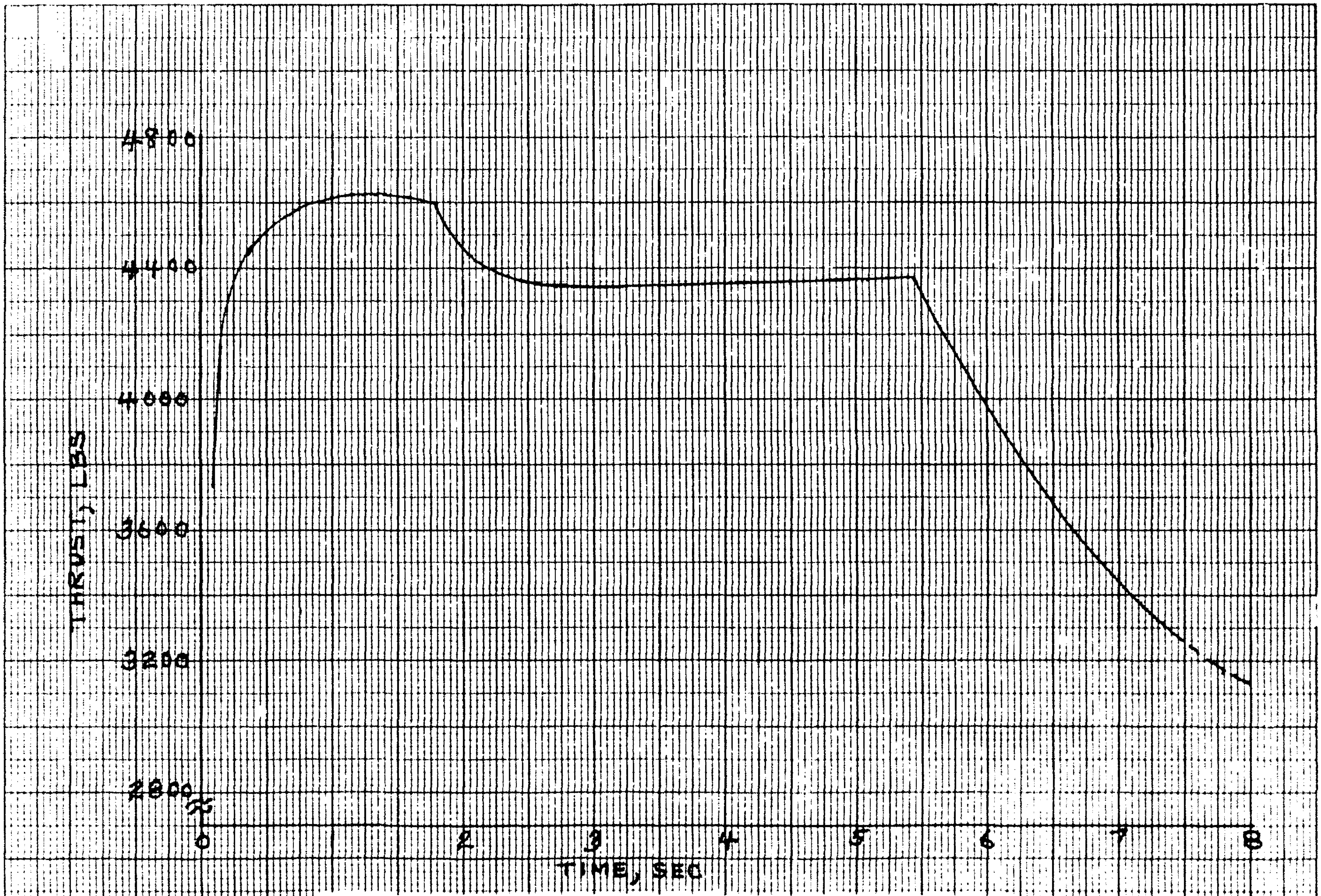

Figure 2.5-9 Nominal Sustainer Engine Full-Throttle Thrust

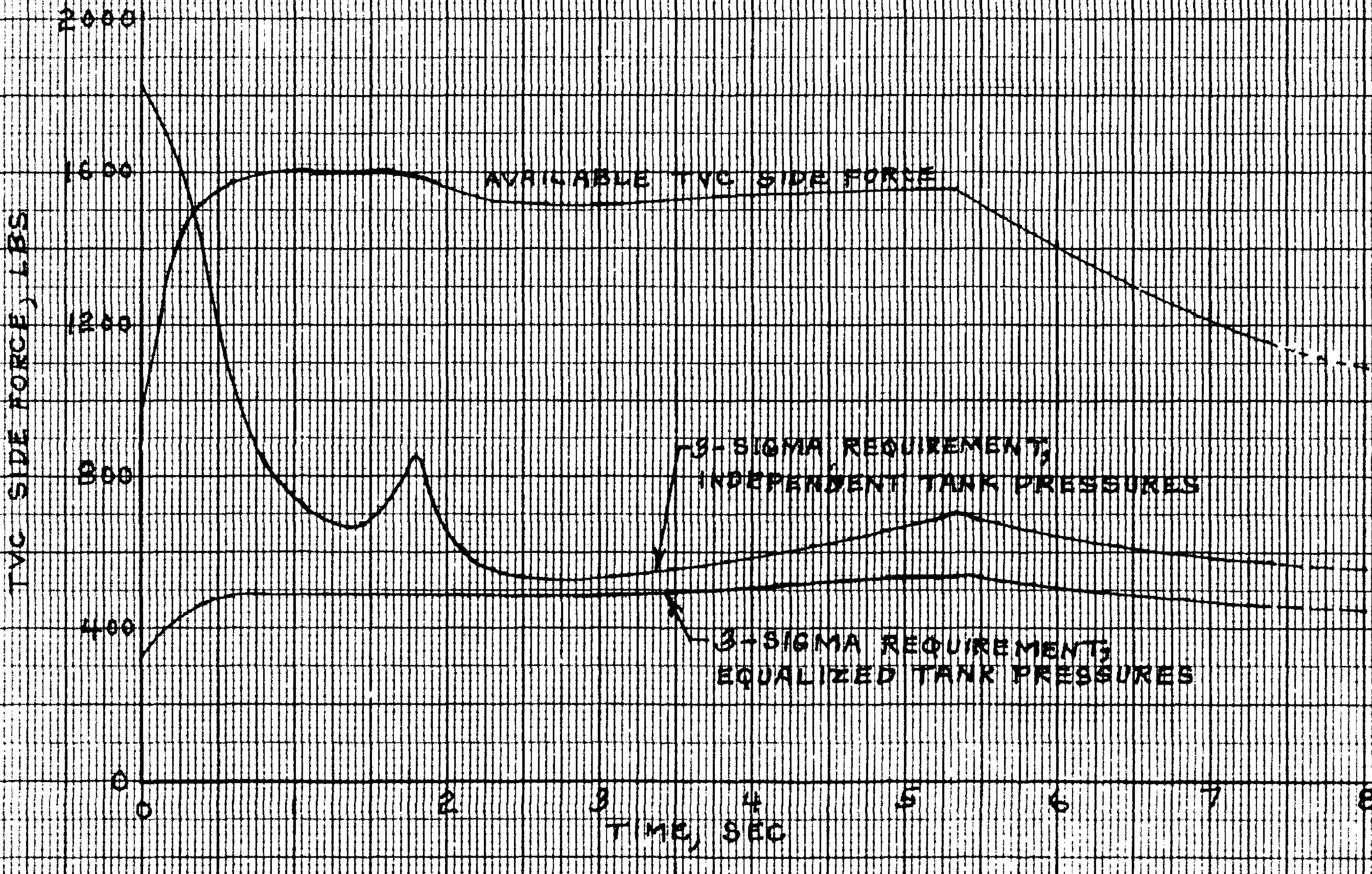

Figure 2.5-10 Required vs. Available Control Authority for CLB Controlled by TVC

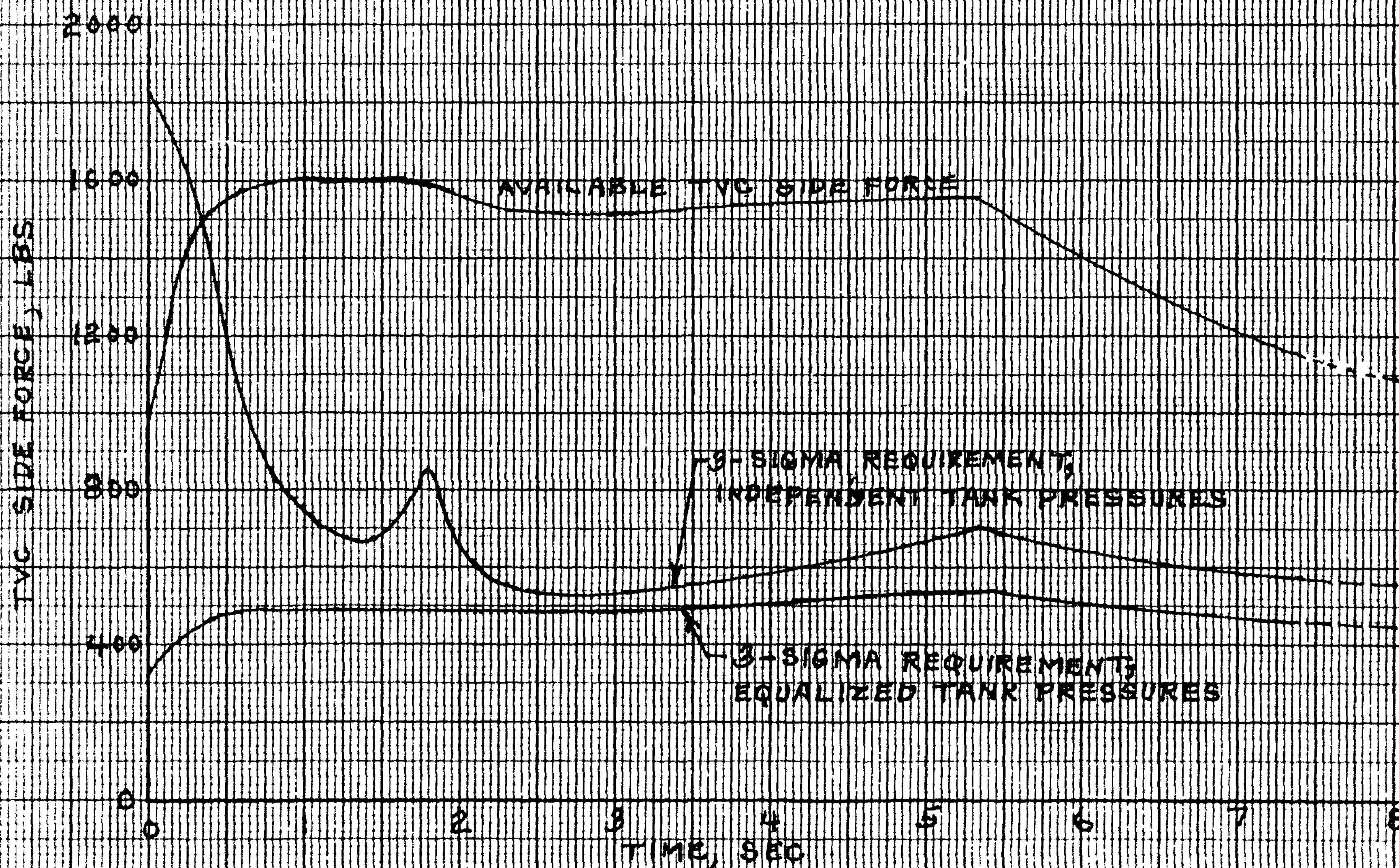

Figure 2.5-10 Required vs. Available Control Authority for CLB Controlled by TVC

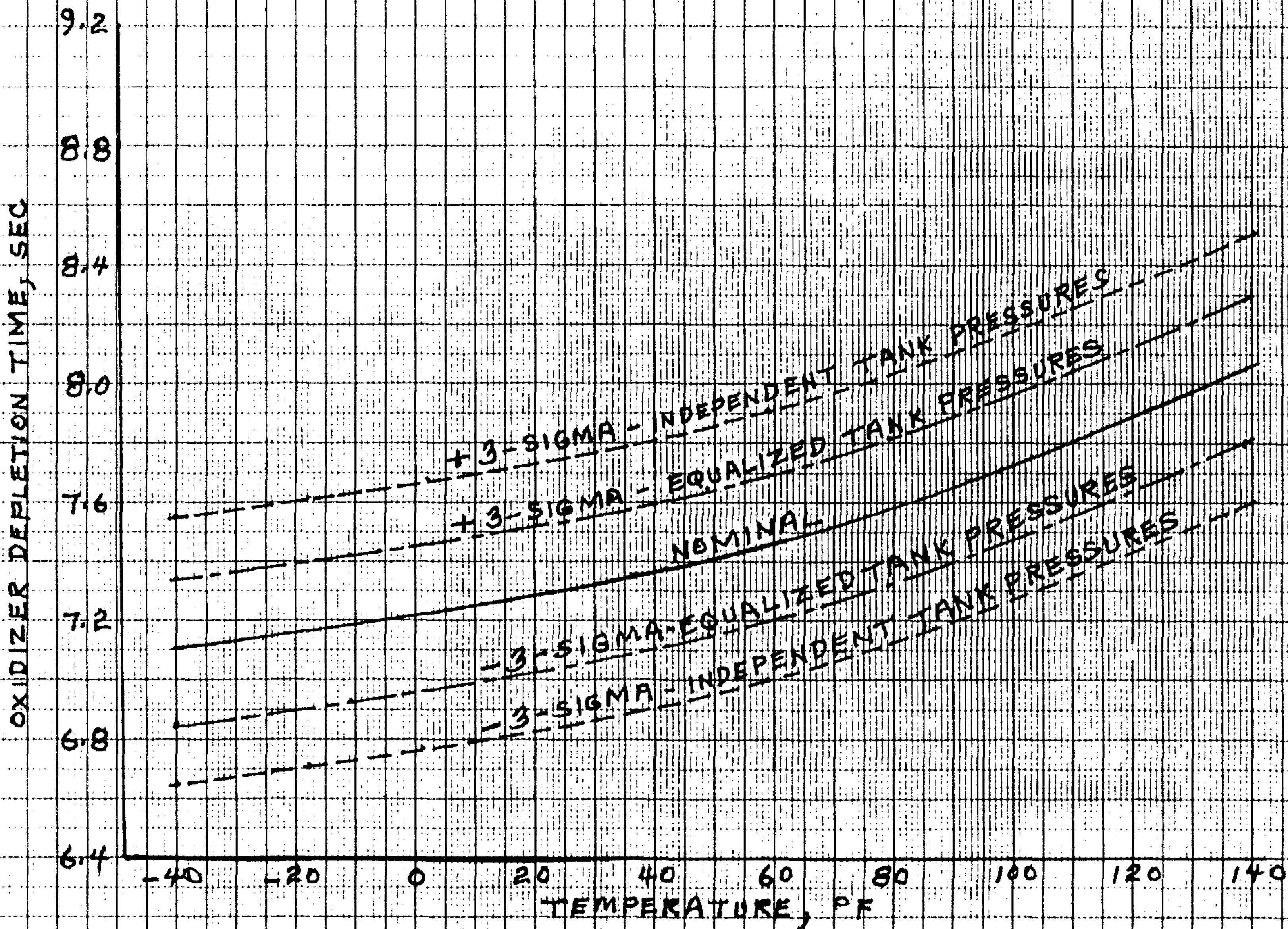

Figure 2.5-12 Estimated Oxidizer Depletion Time

Table 2.5-4

Estimation of Standard Deviation for Oxidizer Depletion Time

ERROR SOURCE	DELTA-T* (sec)	COMMENTS
SPGG pressure	0.127	Derived from BECO errors from flight data with throat area effects removed
Characteristic velocity efficiency	0.054	From engine test data and math model simulation
Engine mixture ratio	0.047	From engine test data and propellant utilization computations
Nozzle throat area	0.029	From engine test data and math model simulation
Engine ignition time	0.023	From static test data
TVC usage	0.009	From math model simulations assuming std dev of 9 lbs for TVC fuel usage
Depletion time std dev for independent tank pressures	0.151	Root-sum-square of all DELTA-Ts
Depletion time std dev for equalized tank pressures	0.081	Root-sum-square of all DELTA-Ts except SPPG pressure DELTA-T

* DELTA-T is the change in oxidizer depletion time caused by one standard deviation of the error source.

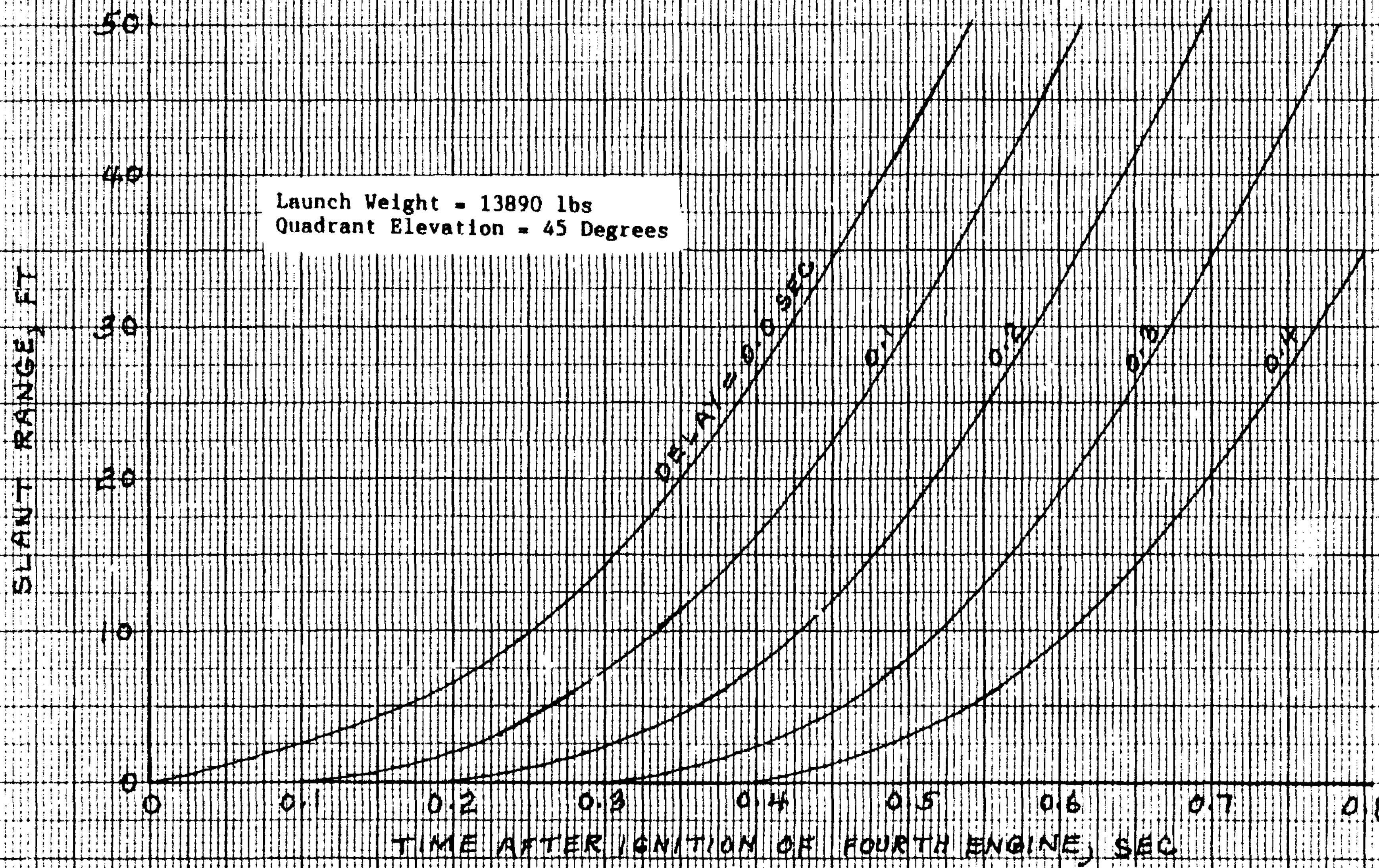

Figure 2.6-1 Required Rail Length for CLB/LPT-40 Vehicle

approximately 0.5 second after engine ignition. Control of sustainer engine thrust can not begin until priming is completed.

Low frequency sustainer chamber pressure oscillations were observed on some Lance flights. The oscillations did not prevent the sustainer from performing its intended function which was to nullify the effects of aerodynamic drag. However, oscillations may not be acceptable for the CLB application. Stability of the sustainer engine control system should be reviewed before selecting sustainer engine thrust control as a means of equalizing the thrust of the CLB propulsion systems.

2.5.3 Required vs. Available Control Authority

Data from the previous sections was used to generate figures 2.5-10 and 2.5-11 which show a comparison of available versus required control authority. Figure 2.5-10 indicates that the TVC system would provide adequate control authority after about 0.35 second. Figure 2.5-11 indicates that sustainer thrust control provides adequate control authority after about 0.5 second.

2.5.4 Estimated Propellant Depletion Time

Thrust of the individual propulsion systems must be terminated in a manner which will not cause large pitch and yaw moments on the vehicle. An estimate of propellant depletion time is required to establish criteria for termination of thrust. Normally, oxidizer is depleted first, unless an excessive amount of fuel is used by the TVC system (see section 2.5.2.1). The analyses which follow are based on oxidizer depleting first.

The average oxidizer depletion time with 3-sigma limits is shown in figure 2.5-12 as a function of system temperature. The average depletion time was computed by the Lance math model for temperatures of -40, 59 and 140 degrees F using the SPGG pressure data from figure 2.5-3. The math model also accounts for the effects of temperature on propellant densities and engine efficiencies. The 3-sigma limits shown in figure 2.5-12 were computed as indicated in table 2.5-4.

2.6 Launch Rail Requirements

A launch rail may be required to guide the vehicle during liftoff. Simulations were performed to determine slant range and velocity during the first second of operation for both the CLB/LFT-40 and CLB/LFT-22 vehicles.

The slant range data for the CLB/LFT-40 vehicle are shown in figure 2.6-1. The data are presented for several delay times. It may be necessary to constrain the vehicle during the ignition transient to allow time for propulsion system operation to stabilize. Delay time is defined as the time interval between ignition of the fourth engine and liftoff.

The slant range for the CLB/LFT-22 vehicle is shown in figure 2.6-2. End-of-rail velocity is shown in figure 2.6-3.

2.7 Attitude Control System

2.7.1 Autopilot Analysis

An analysis of the closed-loop stability and control aspects of the Clustered Lance Booster (CLB) was conducted. Linear analysis was used to establish autopilot gains for the system, then nonlinear simulation was employed to generate time histories for the boost phase of the mission (0 through 8 seconds). System response to disturbances such as tipoff and thrust misalignment were investigated. The CLB was able to rapidly recover from these disturbances and fly to the end of boost with angles-of-attack remaining less than five degrees in the entire supersonic regime. The analysis shows the feasibility of controlling the CLB throughout it's boost flight. The boost regime was the only flight segment considered in this preliminary analysis.

Four areas of configuration data were needed to complete the analyses: aerodynamics, mass properties, thrust, and control characteristics. The aero data were in a process of refinement during this study (primarily directed toward moving the center of pressure location further aft). The normal force coefficient slope was taken with an assumed center of pressure location to produce force and moment derivatives (with respect to angle-of-attack) as a function of Mach number. The stability levels assumed (one- quarter to one-half caliber stable) are achievable, although neutral stability could probably be accommodated. Baseline mass properties and thrust data were used. Previous work indicated that the available control moment from the thrust vector control (TVC) system was about 22,000 N-m (16,000 lb-ft) with a first-order time constant of about 10.0 milliseconds. The configuration data are summarized below.

Using the information above, a model of the CLB was developed in MATRIXx and analyzed, leading to the autopilot configuration shown in figure 2.7-1. The autopilot outer loop controls pitch attitude with a proportional+integral controller; an inner rate feedback loop provides increased airframe damping and bandwidth. This model was analyzed in the frequency domain, and an auto-pilot with a bandwidth of approximately 1-2 radians per second was designed. Using this autopilot configuration, the time domain (simulation) analysis could begin.

The primary results desired from the simulation analyses were: (1), how well the closed-loop system recovers from launch disturbances such as tipoff and thrust misalignment, and (2), what angles-of-attack would be attained during the flyout. Tipoff was modeled by starting the simulations at an altitude just off the launch rail, with a given initial velocity and angular rate (the angular rate corresponding to the tipoff). Thrust misalignment and other transient disturbances (winds, etc.) were modeled as an imposed nose-down moment with an initial value of about 21,000 N-m (14,000 lb-ft) decaying to zero in one

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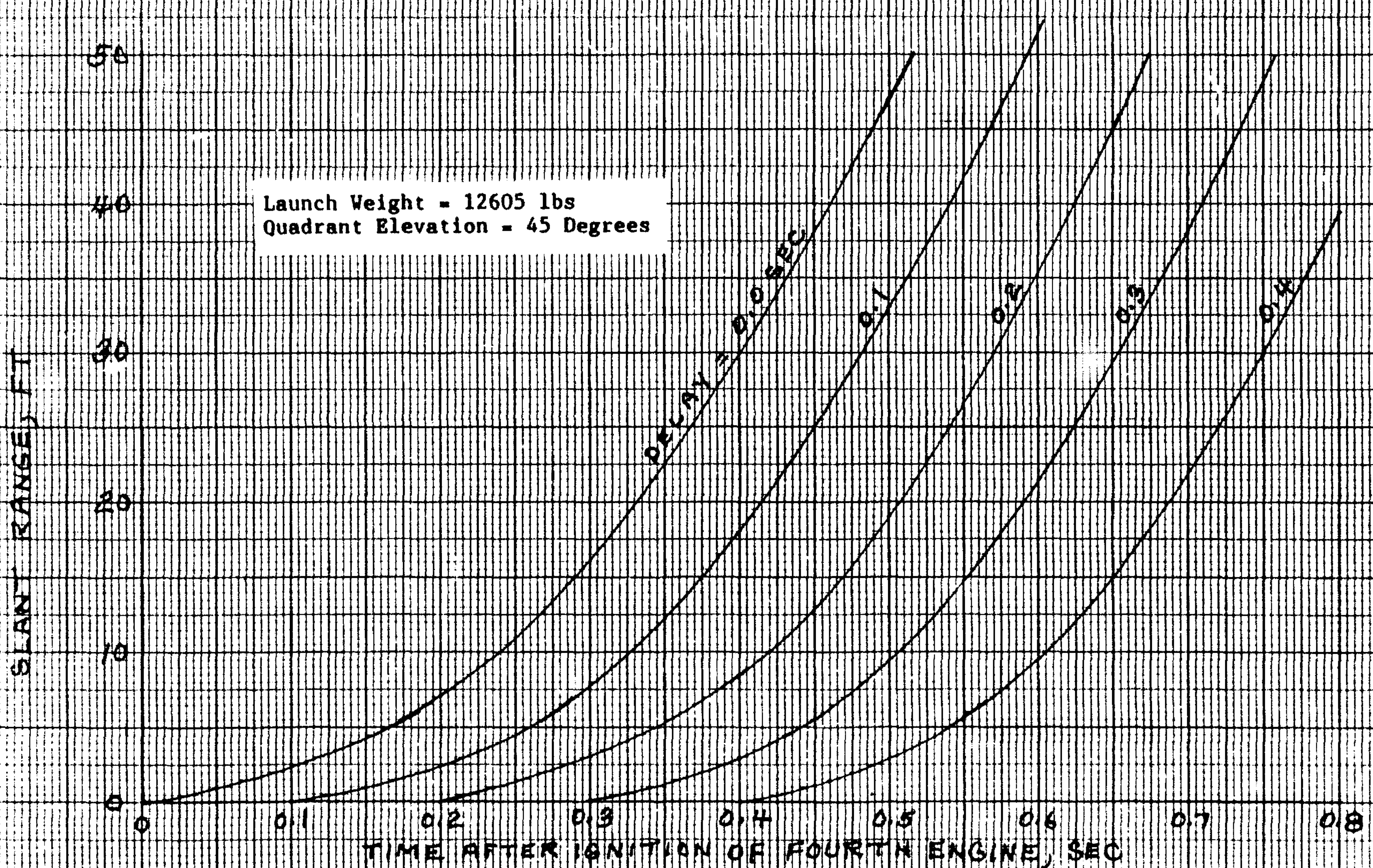


Figure 2.6-2 Required Rail Length for CLB/LFT-22 Vehicle

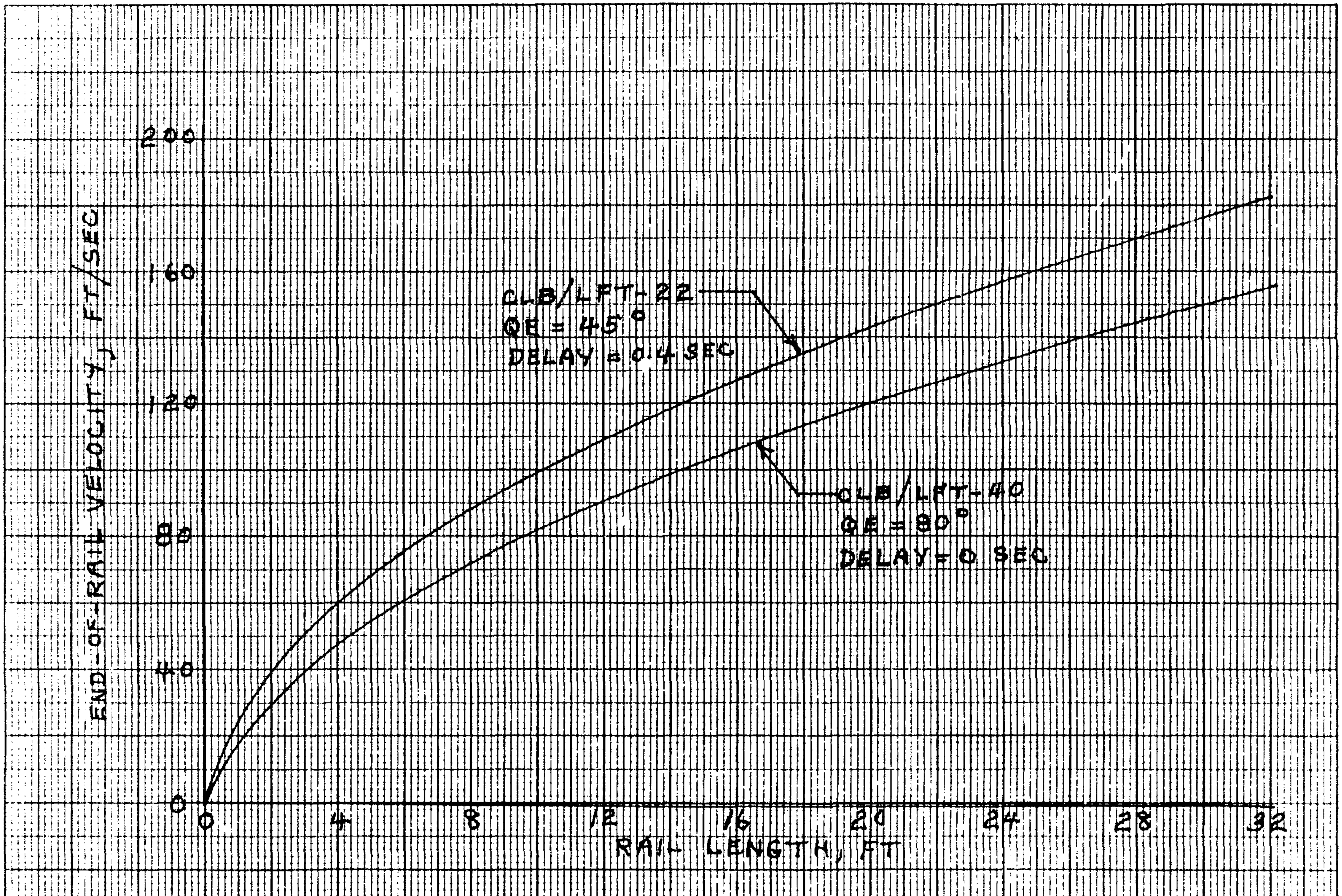


Figure 2.6-3 Estimated End-of-Rail Velocity

- LINEAR ANALYSIS
- ANALYSIS CODE: MATRIXx

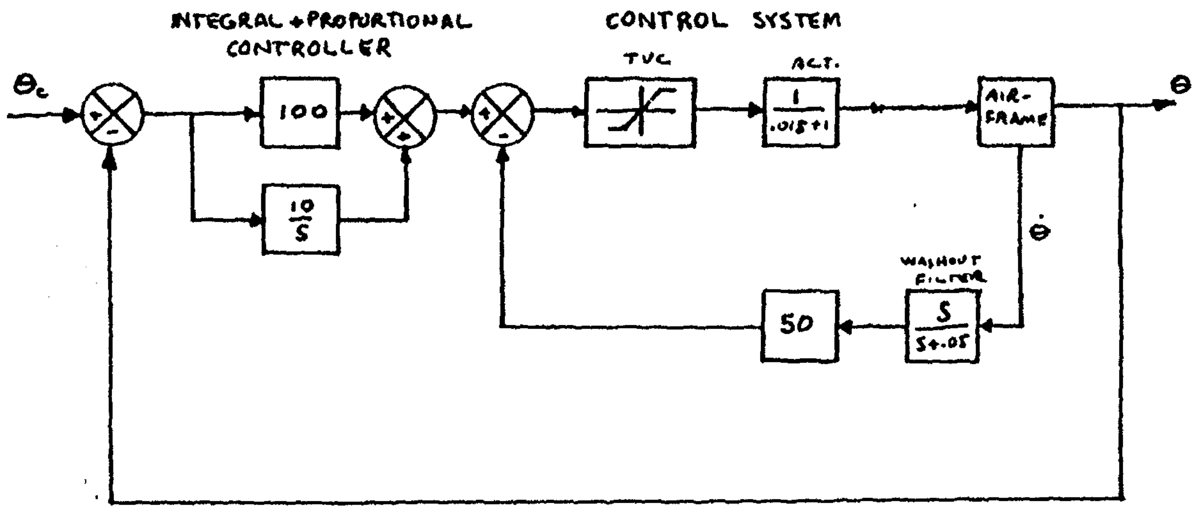


Figure 2.7-1: Clustered Lance Booster Autopilot Configuration

second after launch. The missile was commanded to hold a constant 45.0 degree pitch attitude during the eight second flight. Typical simulation time histories are shown in figures 2.7-2 and 2.7-3. Figure 2.7-2 shows pitch rate, pitch attitude, and angle-of-attack as functions of time. For this extreme case (24 degrees per second tipoff plus the time-varying applied moment), the pitch attitude falls to a value below thirty degrees but has recovered to the commanded level by about three seconds into the flight. The peak angle-of-attack occurs at about 2.5 seconds and reaches about 11 degrees. After 3.5 seconds flight time (Mach number of approximately 1.3), the angle-of-attack remains below 5 degrees. Figure 2.7-3 presents time histories of dynamic pressure, Mach number, and flight velocity for the eight second flight.

In an effort to reduce the pitch attitude variations and maximum angle-of- attack following launch, an additional hot gas reaction control system located in the upper stage was evaluated. A description of this type of control system is presented in section 2.7.2. The revised autopilot concept for this configuration is given in figure 2.7-4. The autopilot block diagram clearly shows the two branches for control (main booster thrust vector control and upper stage hot gas reaction control system). The control improvements achieved with the upper stage reaction control are significant. Figure 2.7-5 clearly shows pitch attitude falling to only 34 degrees (28 degrees previously) and with a maximum angle-of-attack of only 8 degrees (11 degrees previously). Also evident from the figure are the reduced oscillations in both pitch attitude and angle-of-attack. After approximately 3.5 seconds into the flight, the angle-of-attack remained below 4 degrees.

In summary, a short analysis on the controllability of the CLB and the closed- loop response to launch disturbances was conducted. The results clearly show that, for combinations of tipoff, thrust asymmetry, winds, etc., the CLB is controllable. Angle-of-attack, while reaching peaks greater than 5 degrees, remains below 4 degrees after 3.5 seconds of flight.

2.7.2 Attitude Control Concepts

Attitude control is achieved by use of the thrust vector control system located in the four main motor nozzles in the booster. Figure 2.7-6 presents both the available and required control moment (as a function of time for the booster. The required control moment shown in this figure is a result of differential thrust levels of the booster motors that may be present.

The available control moment, using thrust vector control alone, becomes equal to the required value approximately 0.4 second after the fourth motor ignition. Using a 0.2 second booster release delay, six feet of travel would occur before adequate contol authority could be obtained. This would require that a guide rail, six feet in length, be utilized with the launcher, in addition to the hold-back devices). Holding the missile back for longer periods of time would obviously affect the flight performance because of the increased fuel consumption prior to release.

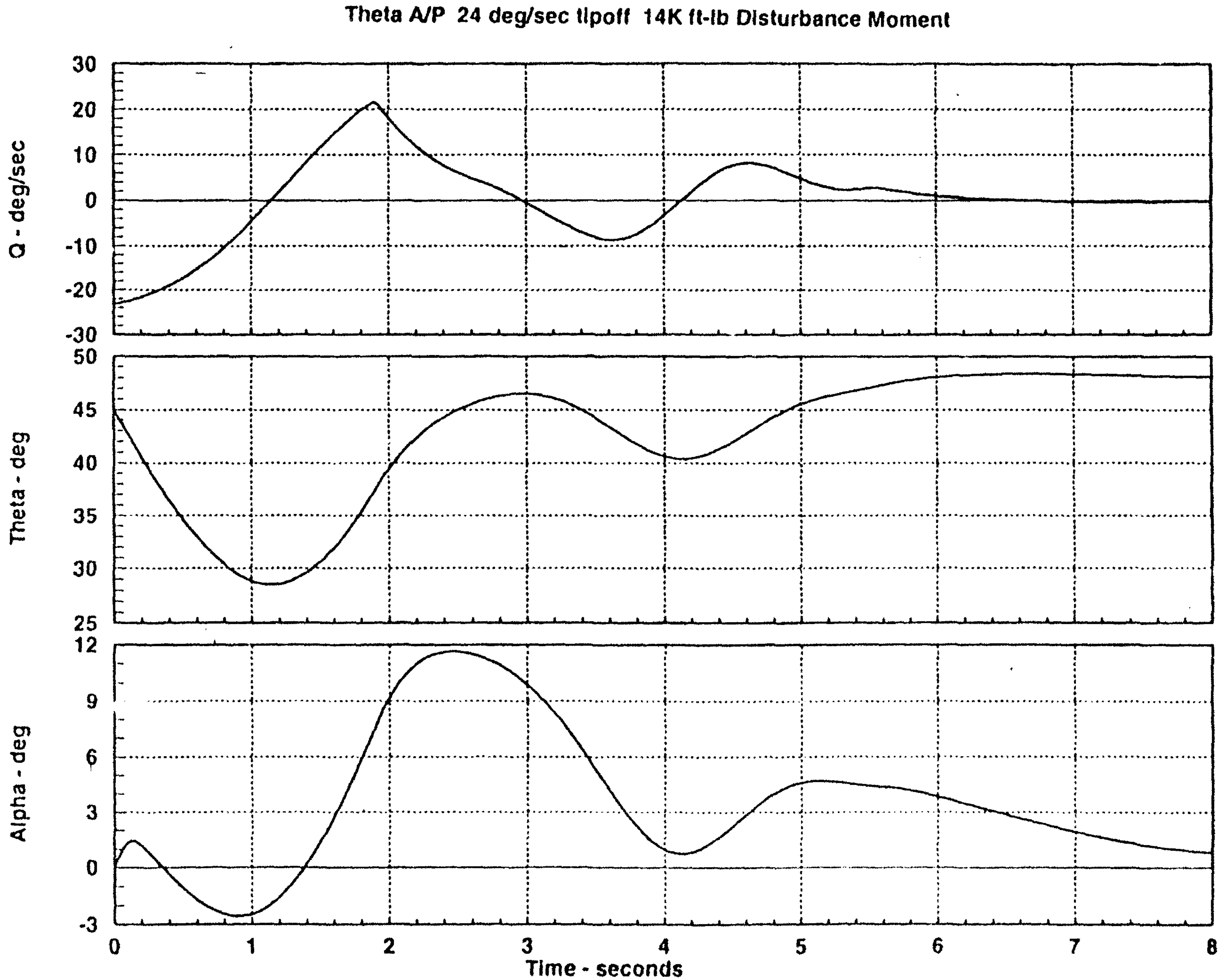


Figure 2.7-2: Airframe Response Using Thrust Vector Control

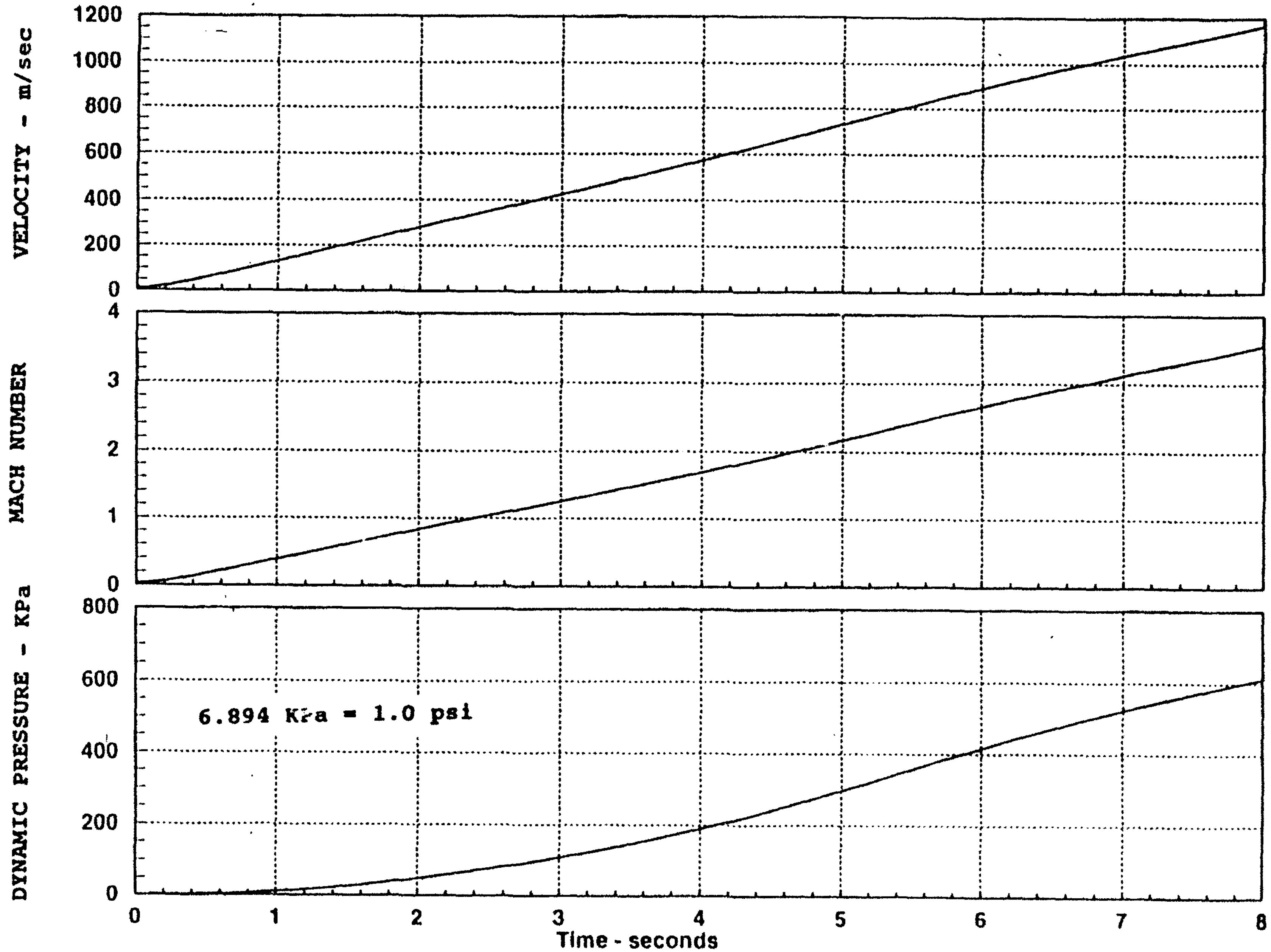


Figure 2.7-3: Clustered Lance Booster Flight Parametrics

- LINEAR ANALYSIS
- ANALYSIS CODE: MATRIXx

UPPER STAGE CONTROL THRUST = 400 POUNDS AT STATION 134.0

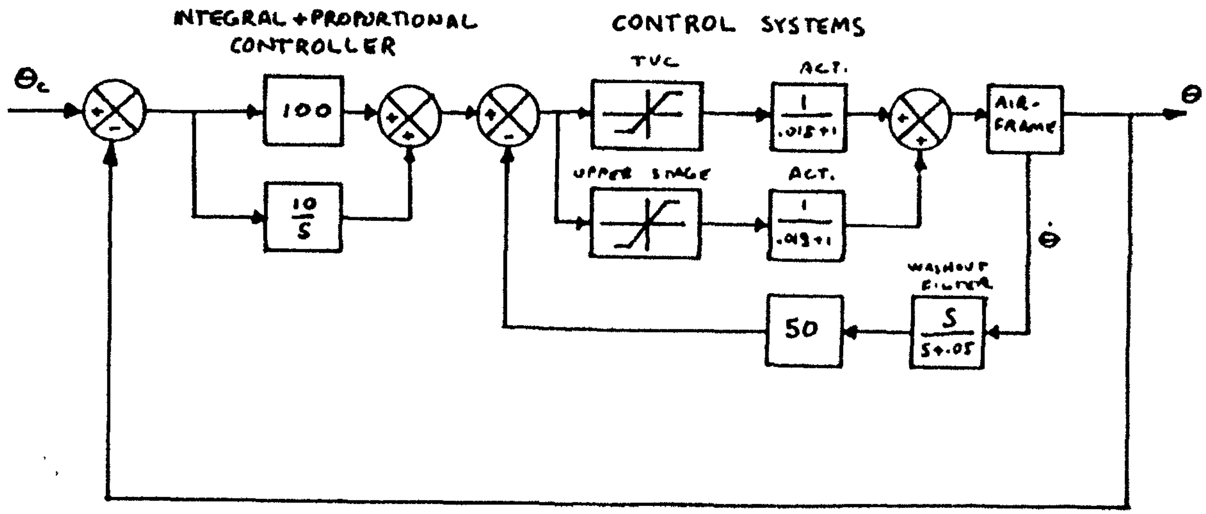


Figure 2.7-4: Clustered Lance Booster Auotpilot Incorporating Upper Stage Reaction Control

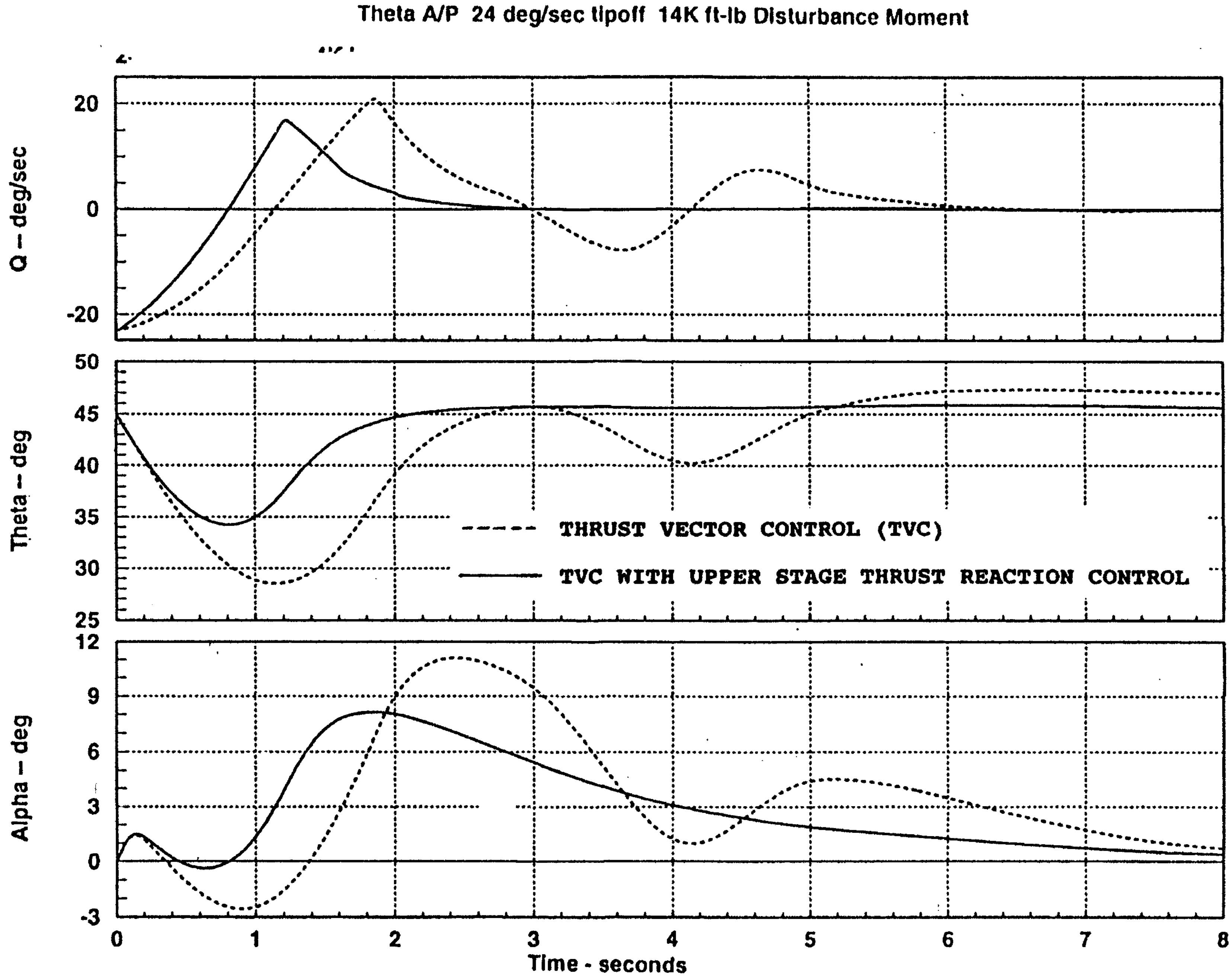


Figure 2.7-5: Airframe Response Using Both Thrust Vector Control and Upper Stage Reaction Control

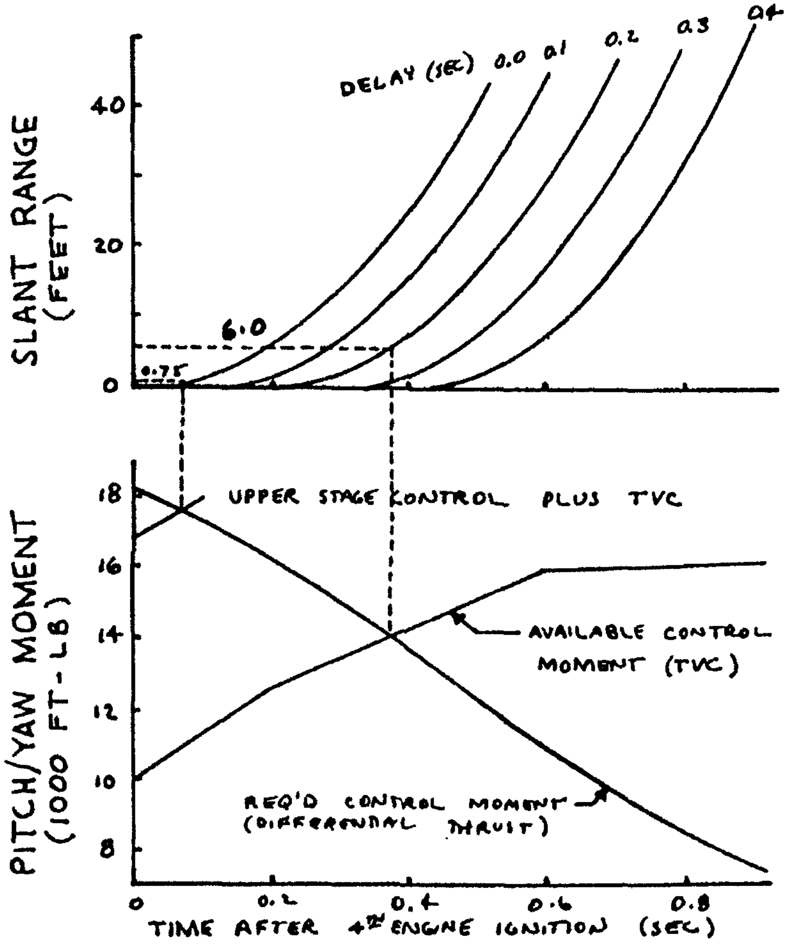


Figure 2.7-6: Clustered Lance Booster Control Parametrics

Additional control authority can be achieved by the use of a hot gas reaction control system located in the upper stage. Figure 2.7-7 shows two different concepts; a liquid propellant system and a pulsed, solid propellant system. Both concepts have 400 pound thrust motors with nozzles located 134.0 inches aft of the nose of the upper stage. The concepts shown both have the capability of providing pitch, yaw and roll control.

The liquid propellant version has the advantage of a demand type propellant usage but suffers because of the complexity/cost involved. The pulsed, solid propellant system is less complex (and less expensive) but it has the disadvantage of consuming propellant while operating; even when no control moment is required. The pulsed feature of the solid propellant control system conserves propellant, but at the cost of limited operation periods. The pulses may be scheduled, for example, during launch, staging, and upper stage guidance phases of the flight.

The available control moment using the upper stage hot gas reaction control plus the main motor thrust vector control is also shown in Figure 2.7-6. A significant reduction of the time required to achieve adequate control is evident. Sufficient control is now available after only 0.1 second after fourth booster motor ignition. This will permit the required launch rail to be less than 1.0 foot in length without any requirements for a booster release delay.

Other control concepts (or combination of concepts) may be considered such as jet vanes located in the rocket motor nozzles and aerodynamic control surfaces. Jet vanes are effective in providing control in both low and high flight speeds. They could, however, produce undesired results with a multiple rocket motor/nozzle configuration where non-uniform thrust characteristics are possible in each nozzle. Aerodynamic controls (controllable tail fins, canards or wings) are effective in the higher velocity regimes, but they provide little control authority at very low speeds. Aerodynamic control actuation also introduces actuator complexities and increased cost.

2.8 Flight Performance

Flight performance of the CLB was estimated by performing three-degree-of-freedom trajectory simulations. Results are shown in figures 2.8-1, 2.8-2 and 2.8-3. The drag coefficients given in section 2.2 were used in the simulations. Vehicle angle of attack was set to zero throughout the flight. A total of six trajectories were generated. Launch weight, second stage weight, launch altitude, first stage burn time and launch angle were varied.

First stage propulsion system thrust and specific impulse used in the simulations are given in figures 2.8-4 and 2.8-5 for sea level conditions. Both thrust and specific impulse are corrected for altitude within the simulation program. First stage performance data are based on Lance test data.

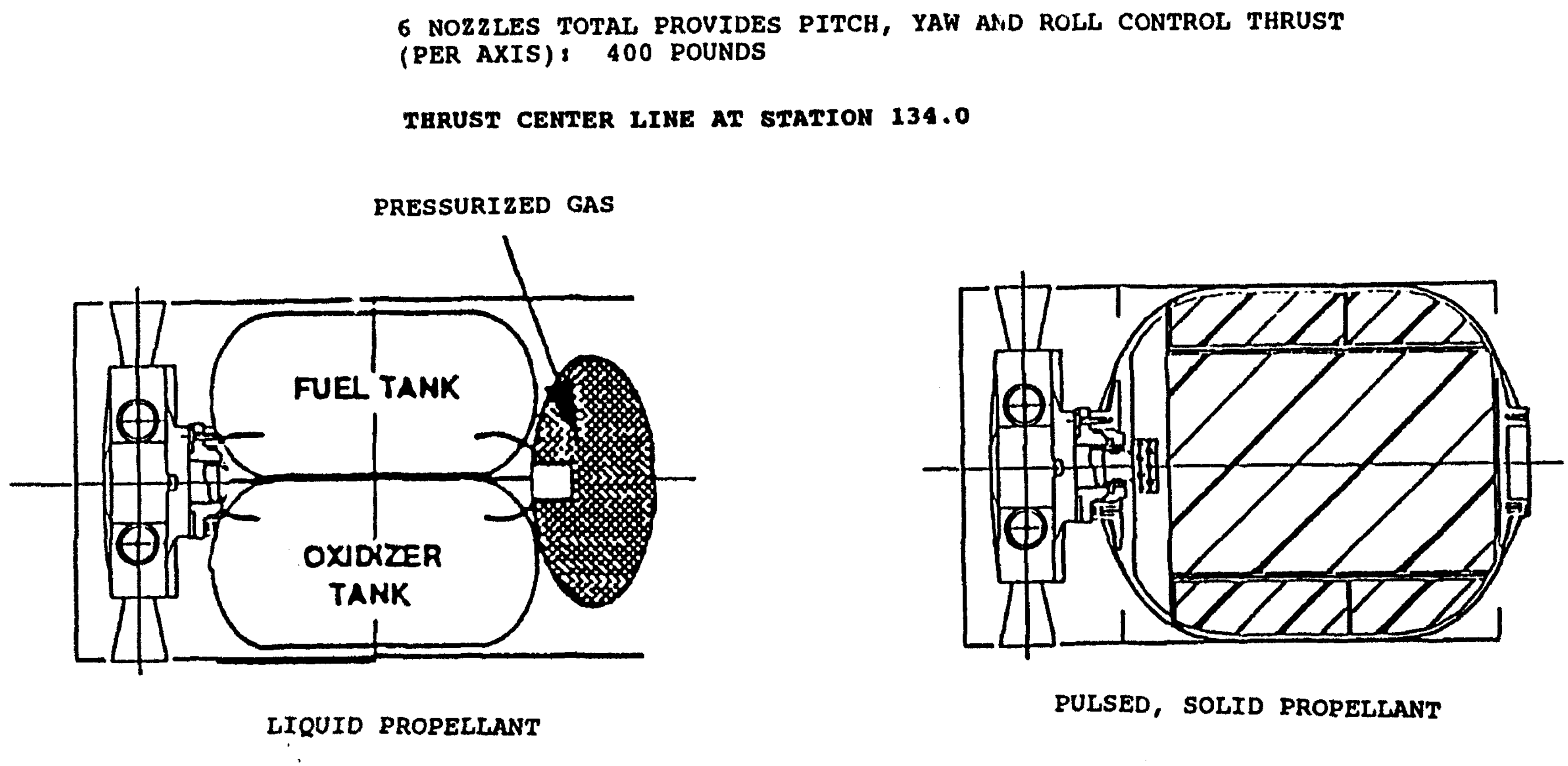
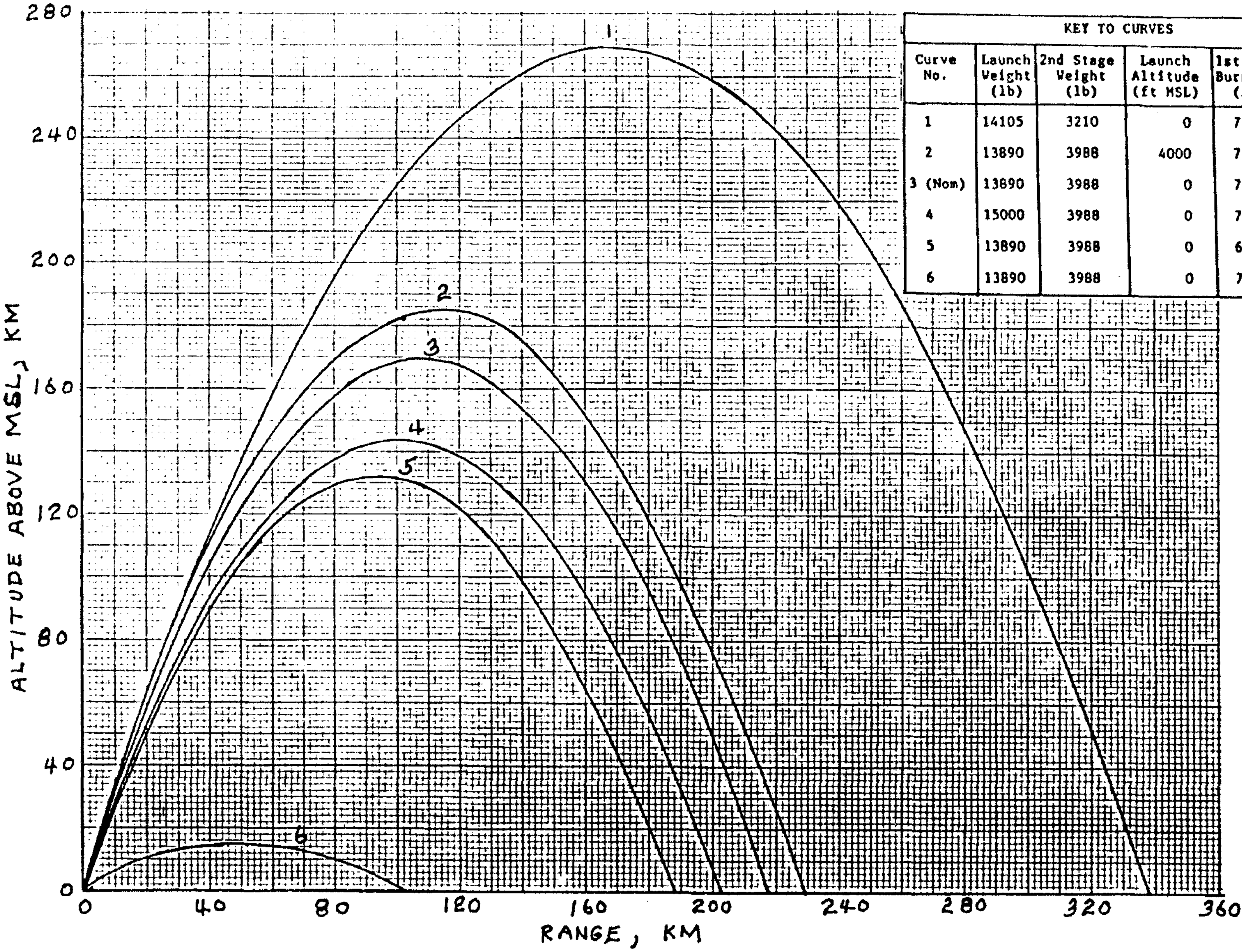
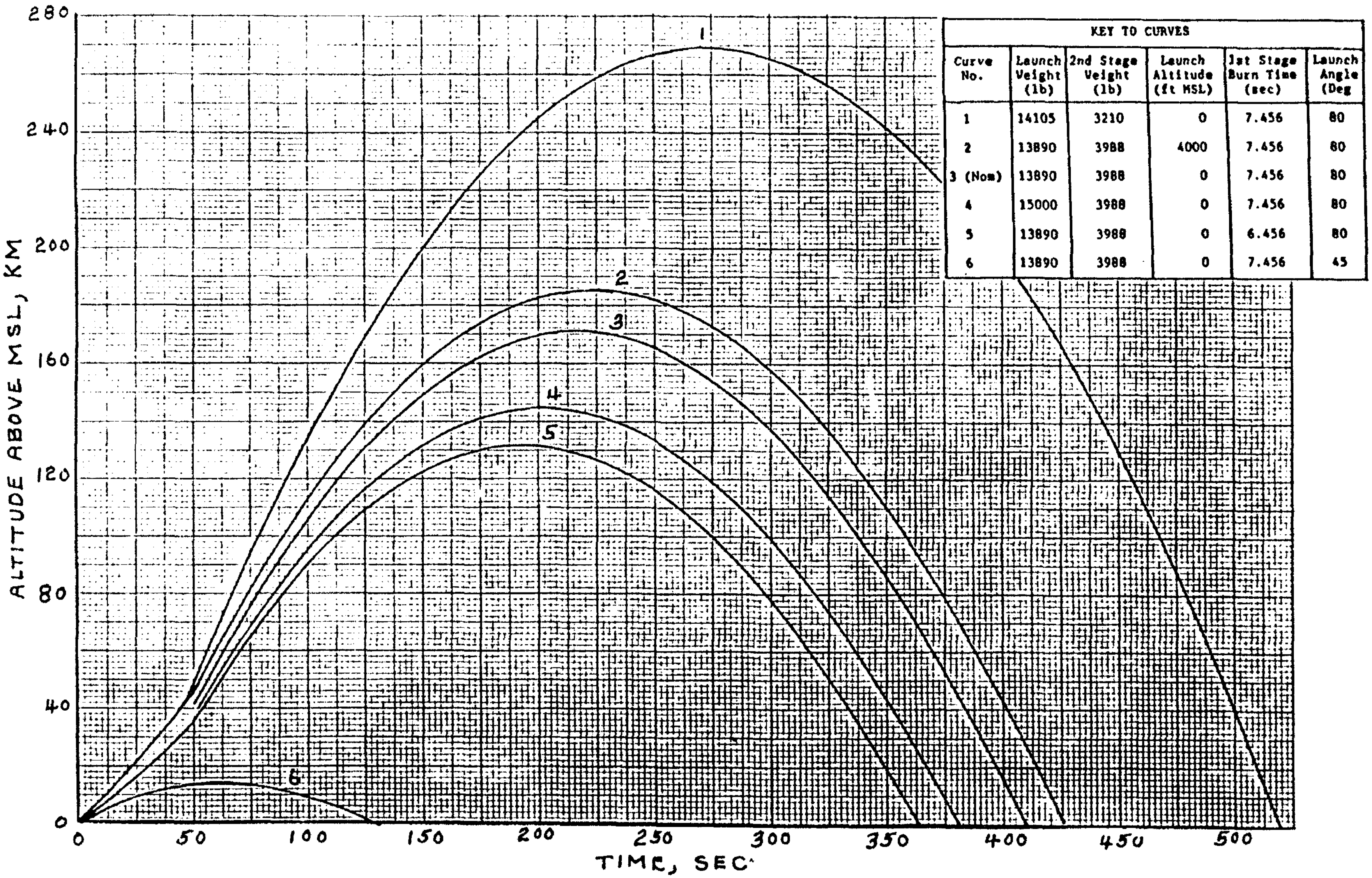


Figure 2.7-7: Upper Stage Control System Schematic



KEY TO CURVES					
Curve No.	Launch Weight (lb)	2nd Stage Weight (lb)	Launch Altitude (ft MSL)	1st Stage Burn Time (sec)	Launch Angle (Deg
1	14105	3210	0	7.456	80
2	13890	3988	4000	7.456	80
3 (Nom)	13890	3988	0	7.456	80
4	15000	3988	0	7.456	80
5	13890	3988	0	6.456	80
6	13890	3988	0	7.456	45

Figure 2.8-1 Estimated Altitude vs. Range



KEY TO CURVES

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1	14105	3210	0	7.456	80
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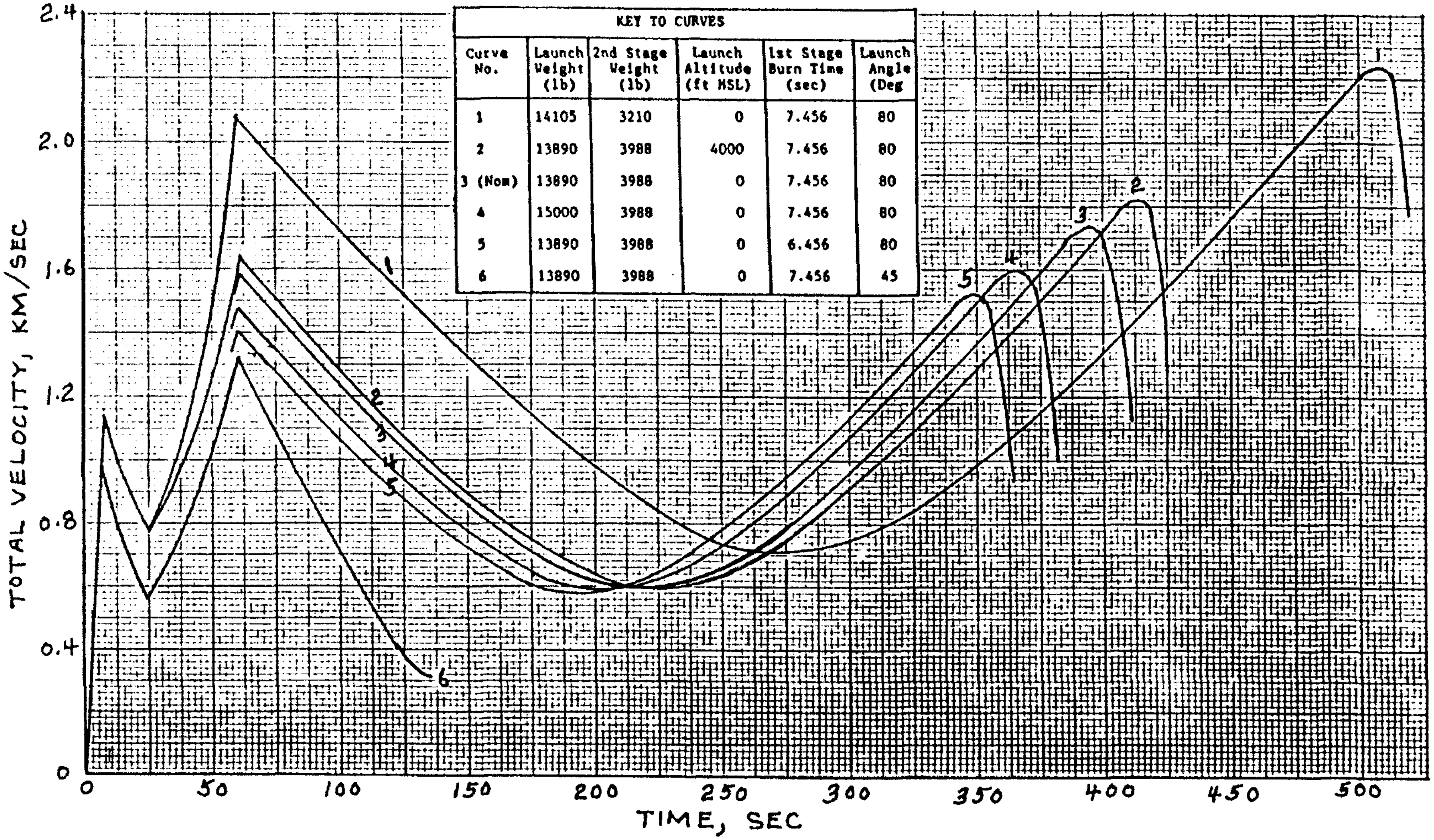
3 (Nom)	13890	3988	0	7.456	80
4	15000	3988	0	7.456	80
5	13890	3988	0	6.456	80
6	13890	3988	0	7.456	45

Figure 2.8-2 Estimated Altitude vs. Time



KEY TO CURVES

Curve No.	Launch Weight (lb)	2nd Stage Weight (lb)	Launch Altitude (ft MSL)	1st Stage Burn Time (sec)	Launch Angle (Deg
1	14105	3210	0	7.456	80
2	13890	3988	4000	7.456	80
3 (Nom)	13890	3988	0	7.456	80
4	15000	3988	0	7.456	80
5	13890	3988	0	6.456	80
6	13890	3988	0	7.456	45

Figure 2.8-3 Estimated Velocity vs. Time

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KEUFFEL & ESSER CO.

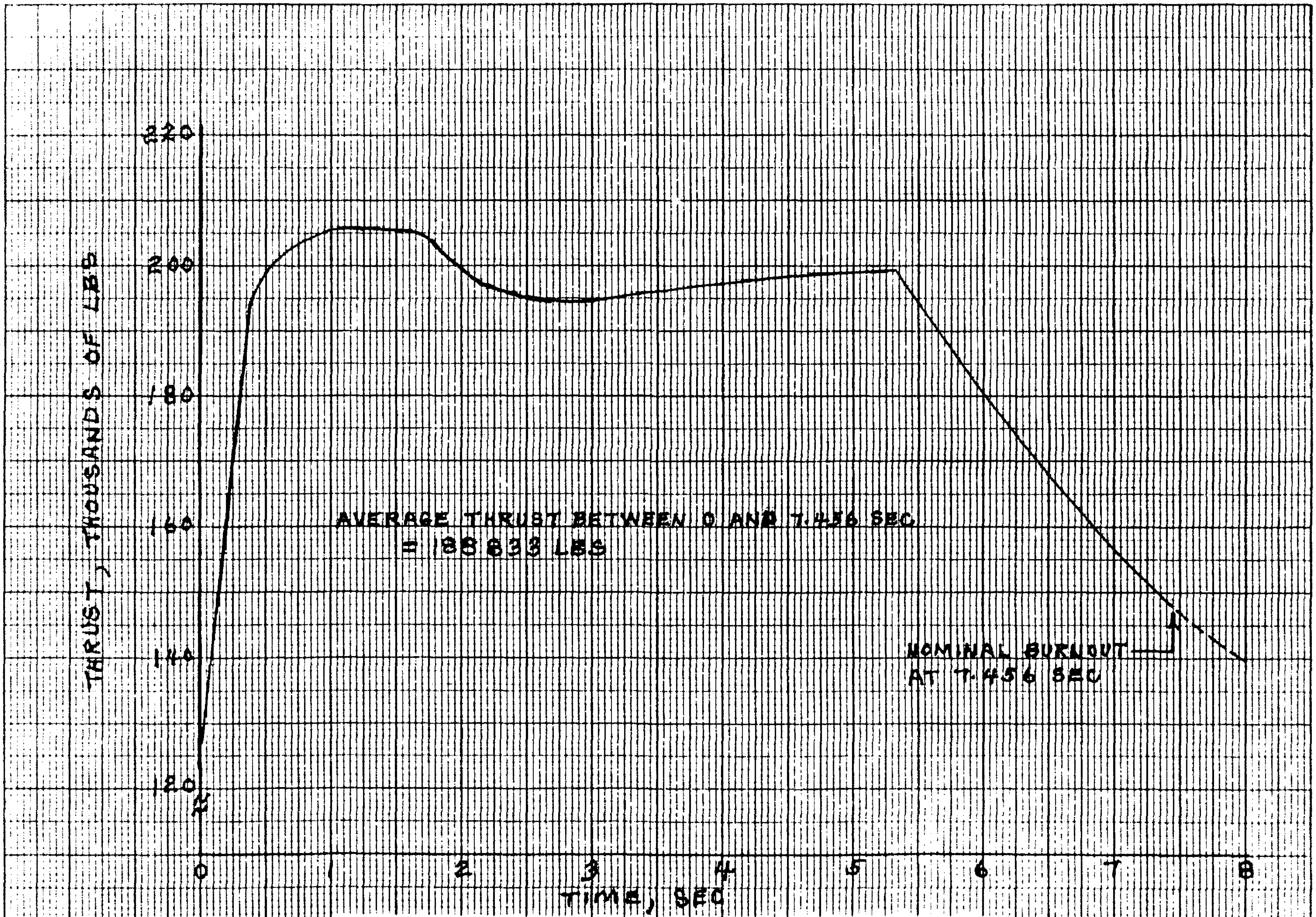

Figure 2.8-4 CLB First Stage Sealevel Thrust

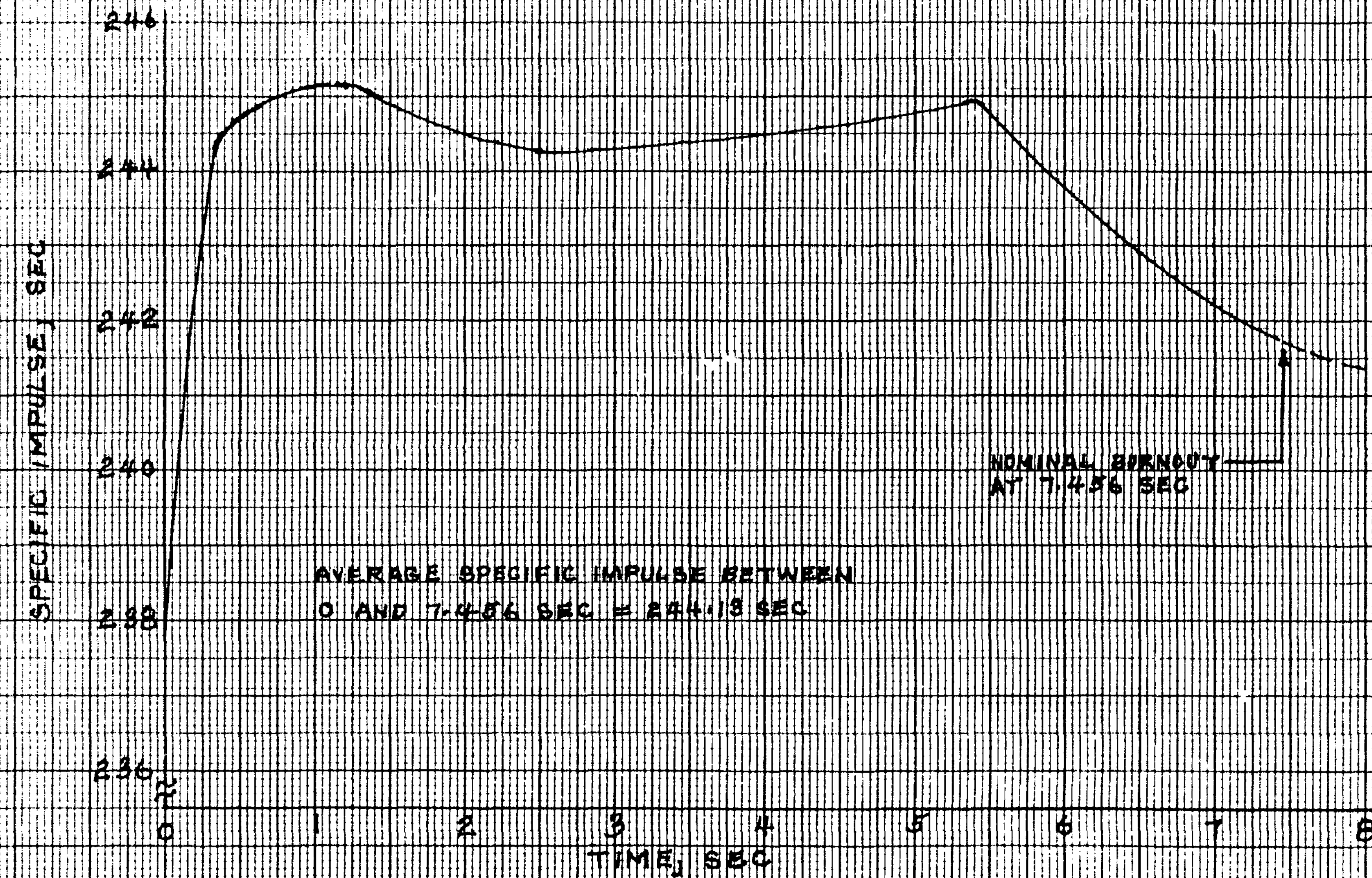

Figure 2.8-5 CLB First Stage Sealevel Specific Impulse

Second stage propulsion system performance was estimated for a Lance propulsion system which was modified to provide a nozzle expansion ratio of 16 and a constant propellant flow rate for 35 seconds. For vacuum conditions, thrust is 11600 pounds, and specific impulse is 278.8 seconds.

Trajectory number 3 is a performance estimate for the CLB/LFT-40 baseline vehicle. Trajectories 2, 4, 5 and 6 are perturbations of CLB/LFT-40 vehicle parameters. Trajectory 2 shows the effects of launch altitude (White Sands Missile Range altitude compared to sea level). Trajectory 4 shows the effect of increasing first stage weight to allow for the additional structure which may be required. Trajectory 5 shows the effects of shortening the boost time, which would be necessary if booster thrust is terminated by times rather than propellant depletion sensors. Trajectory 6 shows the effects of lowering the launch angle from 80 degrees to 45 degrees.

Trajectory number 1 is an estimate of the performance of the CLB/LFT-22 vehicle. First stage weight was increased by 1000 pounds over the baseline wight given in section 2.4, and second stage weight was increased by 500 pounds. The increased performance of the CLB/LFT-22 vehicle is primarily due to the lower weight and drag of the 22-inch diameter second stage when compared to 40-inch diameter second stage of the CLB/LFT-40 vehicle.

FOLLOW-ON STUDIES

One of the objectives of this study was to identify areas requiring additional study before initiating full scale development of the CLB vehicle. A statement of work for the required follow-on studies is given in appendix C. A rough order of magnitude (ROM) estimate of the cost to perform the follow-on studies will be provided under separate cover.

APPENDIX A

CONTRACT STATEMENT OF WORK

U. S. Army Strategic Defense Command
P.O. Box 1500
Huntsville, Alabama 35807-3801

SCOPE OF WORK

SW-D-56-92
29 Jun 92
Revision 1

CLUSTERED LANCE BOOSTER EVALUATION

1.0 BACKGROUND. The U.S. Army Strategic Defense Command has a requirement for a liquid propellant, first-stage booster propulsion system. A thirteen week concept feasibility evaluation of the modification and fabrication of four Lance propulsion systems into a clustered booster configuration is desired. The Lance propulsion system is a candidate for modification to the USASDC requirements for the following reasons.

1.1. The Lance propulsion system is a storable hypergolic, and can be shipped, handled and launched as a "wooden round." There will be no requirement to supply or load propellants at the launch site. This minimizes range safety and environmental impact problems compared to other liquid propellant systems.

1.2. The Lance system is currently being decommissioned and demilitarized/destroyed at a cost to the U.S. Government. Therefore, Lance propulsion systems are available in quantity for such an application.

2.0 OBJECTIVE. To determine the feasibility of developing the conversion of tactical U.S. Army Lance propulsion systems into a high thrust boost stage for U.S. Army Strategic Defense Command use.

3.0 TECHNICAL REQUIREMENTS/TASKS.

3.1. The contractor shall investigate the concept feasibility of clustering four tactical Lance propulsion systems (missile main assemblage) into a high thrust boost stage configuration. The contractor shall consider the following in the performance of this evaluation.

3.1.1. Maximum thrust and burntime from the boost stage is desired.

3.1.2. An unguided and stable boost stage is desired through boost stage burnout.

3.1.3. The spin nozzles shall be removed or disabled.

3.1.4. All components and hardware not required for this application shall be removed to reduce stage weight.

3.1.5. The clustered booster shall be rail launched at 45 to 80 degrees quadrant elevation.

3.1.6. Assembly of the clustered booster configuration at the launch site assembly building is desired.

3.2. The following assumptions are available to the contractor.

3.2.1. Electrical interface with the upper stage will provide simultaneous engine ignition and simultaneous thrust termination signals.

3.2.2. An upper stage guidance, navigation and control section will provide all necessary inputs to the clustered boost stage. (See Figure 1.)

3.2.3. The clustered boost stage engines will perform within the normal Lance engine statistical (1 sigma) limits.

3.2.4. The guidance, navigation and control section for a 22 inch diameter Liquid Fueled Target upper stage will be available, will fit within the Lance warhead envelope, and will have the same mass properties as the current light and heavy Lance warhead. (See Figure 2.)

3.3. The contractor shall perform the following analysis.

3.3.1. Definition of a conceptual design.

3.3.2. Preliminary aerodynamic loads.

3.3.3. Preliminary structural loads.

3.3.4. Structural and stress analysis, including weight analysis.

3.3.5. Preliminary structural integrity assessment.

3.3.6. Projected flight performance assuming a Liquid Fueled Target second stage (see enclosed figures and tables for description of possible configurations).

3.1.2. An unguided and stable boost stage is desired through boost stage burnout.

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3.3.4. Structural and stress analysis, including weight analysis.

3.3.5. Preliminary structural integrity assessment.

3.3.6. Projected flight performance assuming a Liquid Fueled Target second stage (see enclosed figures and tables for description of possible configurations).

3.3.7. Determine if control authority (pitch, yaw and roll) during powered flight of the clustered Lance booster is required.

3.4. The contractor shall, after completing the tasks of paragraph 3.3, prepare a report on the feasibility of proceeding with the development of the clustered booster concept. If deemed feasible, the contractor shall identify those areas requiring additional study, under separate contract, which must be resolved to create a requirements document which will identify all expected Lance propulsion system modifications, testing/handling requirements and a realistic schedule for design, development, test and delivery program.

3.5. The contractor shall identify all projected costs, in FY92 dollars, to conduct additional studies necessary to develop a requirements document and to define a development program in sufficient depth so that total program costs can be reasonably estimated.

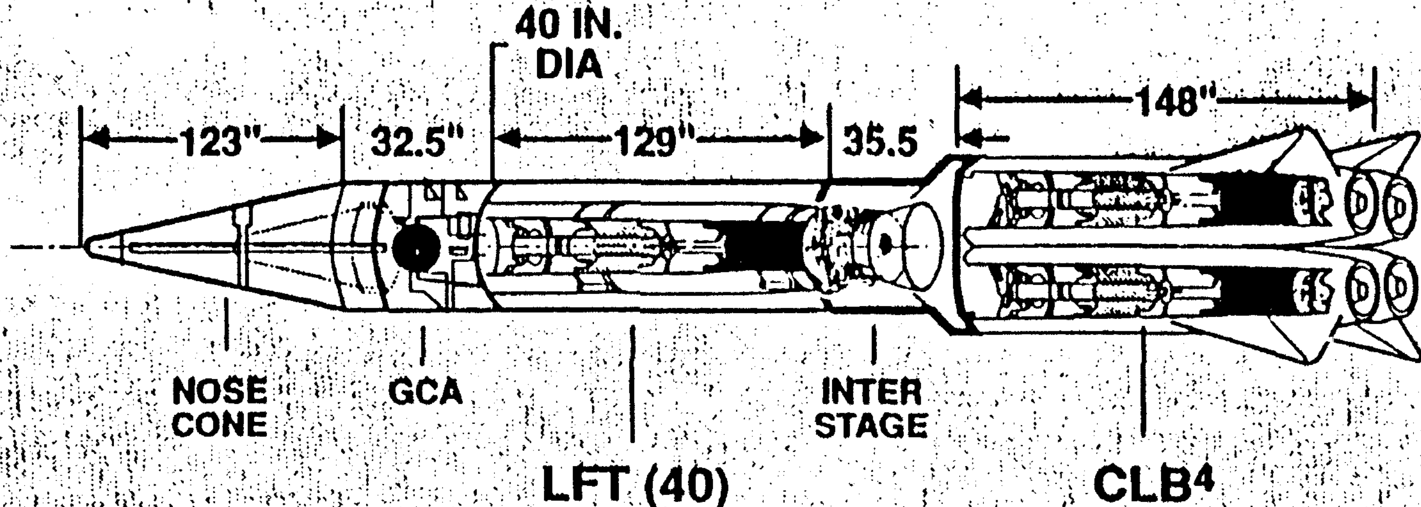
4.0 REPORTING/DATA REQUIREMENTS. The delivery date for the final report under this contract shall be 90 calendar days after contract award and shall be IAW Data Item Description DI-MISC-80048.

UNCLASSIFIED

LIQUID FUELED TARGET (40") / CLUSTERED LANCE BOOSTER (CLB^4) (U)

M-920313-09U (C) (2076)

LFT (40) / CLB^4



NOTE: CLUSTERED LANCE BOOSTER (CLB)-MODIFIED TO PRODUCE BOOST THRUST (APPX. 200K) FOR TOTAL BURN (APPX. 7 SECONDS).
300/400 LB. THRUSTERS FOR PITCH/YAW CONTROL OPTIONAL. COLD GAS ROLL CONTROL

UNCLASSIFIED

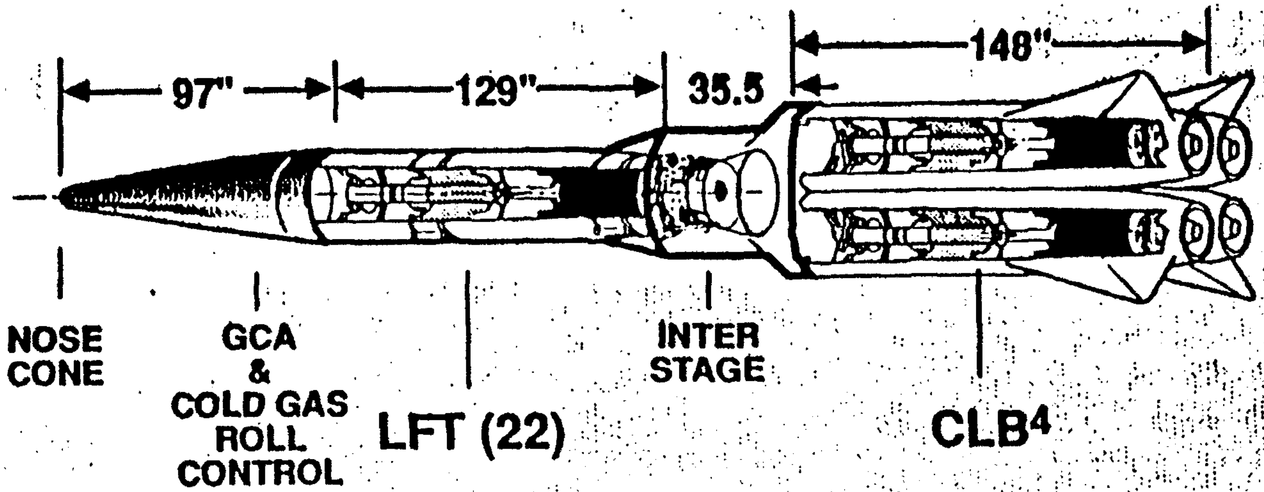
UNCLASSIFIED

UNCl SSIFIED

LIQUID FUELED TARGET (22") / CLUSTERED LANCE BOOSTER (CLB^4) (U)

M-920313-08U (C) (2076)

LFT (22) / CLB^4



NOTE: CLUSTERED LANCE BOOSTER (CLB)-MODIFIED TO PRODUCE BOOST THRUST (APPX. 200K) FOR TOTAL BURN (APPX. 7 SECONDS).
300/400 LB. THRUSTERS FOR PITCH/YAW CONTROL OPTIONAL.
COLD GAS ROLL CONTROL

UNCLASSIFIED

UNCLASSIFIED

ESTIMATED MASS PROPERTIES
(weights in pounds)

LFT (22") / CLB4

LFT (22")	2233
Shroud	465
GCA	500
Interstage	492
CLB4	9000
Structure	700
TOTAL	13390

LFT (40") / CLB4

LFT (40")	2846
Shroud	244
GCA	667
Interstage	492
CLB4	9000
Structure	700
Payload	137
TOTAL	14086
Expected Wt. Savings	200
	13886

TABLE 1

APPENDIX B

LANCE FIELD ARTILLERY MISSILE SYSTEM

FOREWORD

This brochure provides physical and operational characteristics of the LANCE Missile System and a brief description of key components that may be reused in other applications when the system is retired from active service.

LANCE is a reliable, low-cost guided missile system that can be operated under any climatic conditions in which infantry, armored, mechanized, or airborne divisions may be committed. The complete tactical missile system consists of the LANCE missile, two types of launchers, a loader-transporter vehicle, and ancillary equipment. The missile incorporates a single-stage, prepackaged, liquid-propellant rocket and an advanced guidance system that is invulnerable to all known electronic countermeasures. It is the first Army missile to employ packaged and storable liquid propellants, providing maintainability comparable to that of solid-propellant systems. All LANCE Missile System components are air transportable, and the support vehicles are designed for high mobility on the ground. Its mobility, reliability, and invulnerability to electronic countermeasures combine to make LANCE particularly valuable in close support of rapidly moving troops.

Inquiries regarding reuse should be made to:

Commander
U.S. Army Missile Command
ATTN: AMSMI-WS-RS
Redstone Arsenal, AL 35898-5690

Inquires regarding technical matters should be made to:

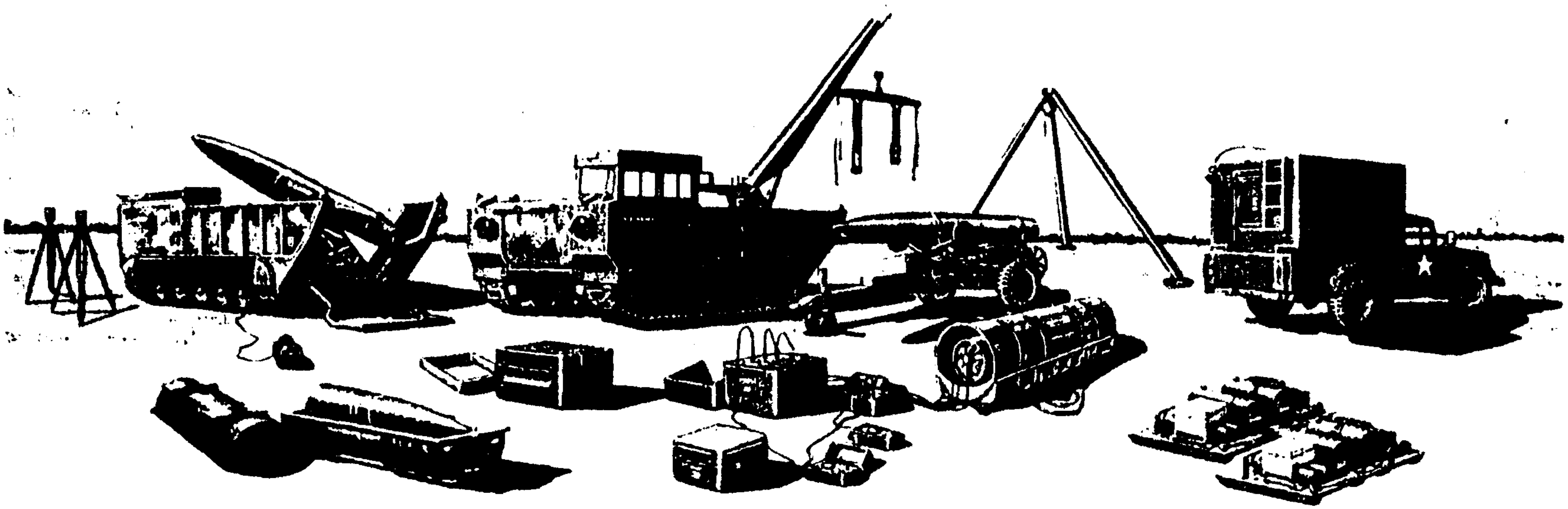
Commander
U.S. Army Missile Command
ATTN: AMSMI-WS-C-EC
Redstone Arsenal, AL 35898-5278

BROCHURE REVISED OCTOBER 1991

LANCE FIELD ARTILLERY MISSILE SYSTEM

TABLE OF CONTENTS

MAJOR COMPONENTS



AZIMUTH LAYING EQUIPMENT

LAUNCHER, CARRIER MOUNTED

FIRING DEVICE

SHIPPING AND STORAGE CONTAINERS

MONITOR PROGRAMMER

LOADER TRANSPORTER

ELECTRONIC TEST SET

LAUNCHER ZERO LENGTH

HOISTING UNIT TRIPOD

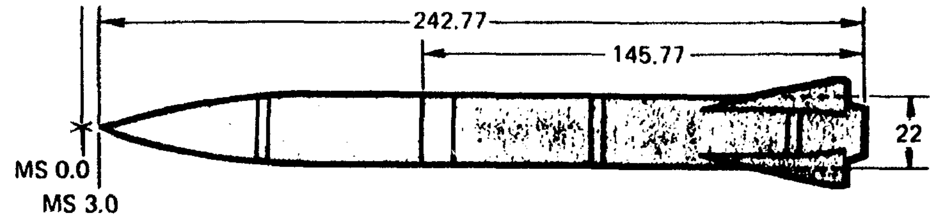
SHELTER, ELECTRICAL EQUIPMENT

PROPELLANT DRAINING KIT

LANCE MISSILE: GM-52C

GENERAL CHARACTERISTICS

Length: 245.77 in. (from MS 0.0)
Propulsion system: bi-propellant liquid; storable
Guidance system: self-contained; directional and velocity control

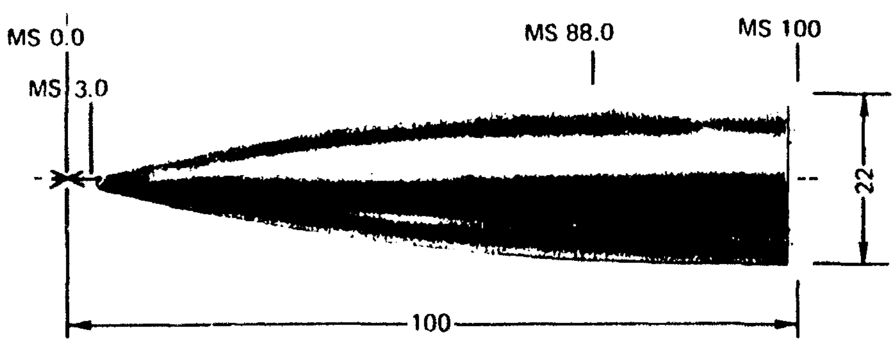


DIMENSIONS IN INCHES

MASS PROPERTIES

Item		Weight (lb)
Missile Round	with heavy warhead section (launch)	3367
	with light warhead section (launch)	2855
	less warhead section and control surfaces	2309
Propellants	Fuel, UDMH	375.5
	Oxidizer, IRFNA	1106.5
Control Surfaces	Large honeycomb – M29 (light warhead section), 4	77
	Small forged aluminum – M30 (heavy warhead section), 4	63

WARHEAD SECTION



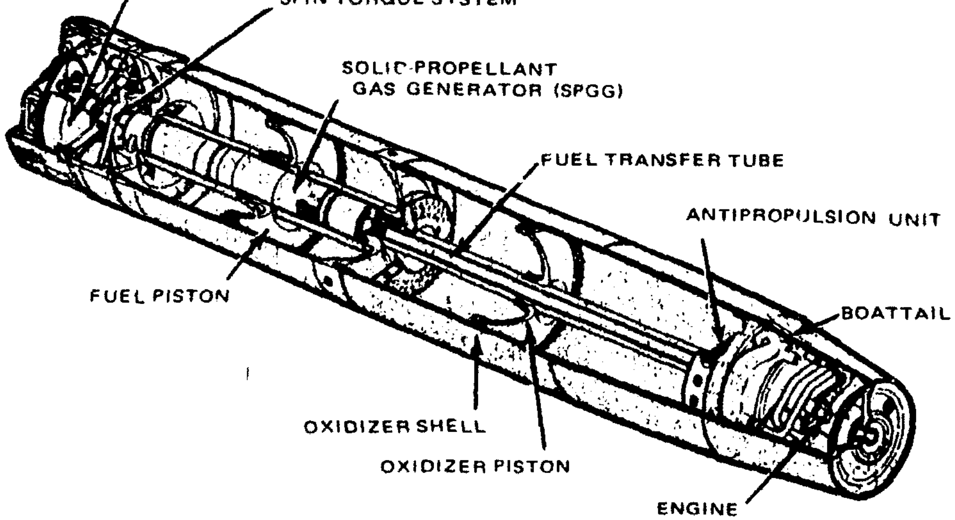
Length: 100 in. from MS 0.0 (97 in. actual)
Maximum diameter: 22 in., between MS 88.0 and 100.0

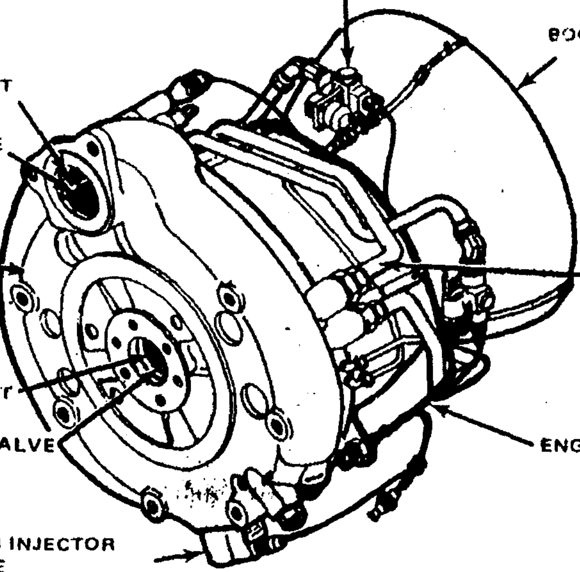
Types:
- M251A1 HE Heavy (Tactical) (995 lb)
- M252 Practice, Light (469 lb)
- XM276 Practice, Heavy (995 lb)

PROPULSION SYSTEM

LANCE's propulsion system – a prepackaged, bi-propellant, liquid-rocket system using unsymmetrical dimethylhydrazine (UDMH) as fuel and inhibited red fuming nitric acid (IRFNA) as oxidizer – consists of the following major subsystems:

- Dual-thrust-chamber engine system
- Propellant-feed system
- Spin system
- Thrust-vector-control (TVC) system



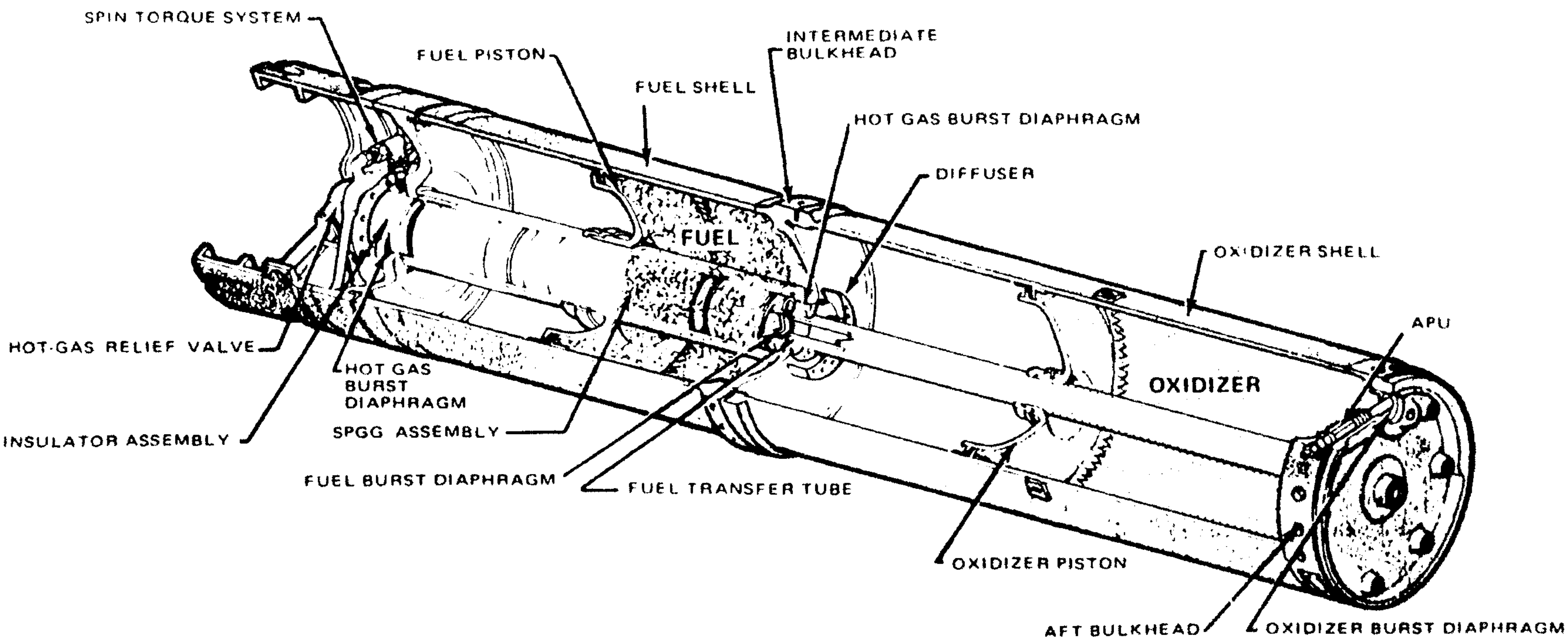


ENGINE SYSTEM

In the dual-thrust-chamber engine, the annular thrust chamber of the booster engine surrounds, and is coaxial with that of the sustainer engine. Both engines operate during the boost phase of flight. The fixed-thrust booster has a variable-cutoff capability that permits selection of any boost phase duration between the minimum and maximum system capabilities, thus providing the necessary range control. Booster thrust is terminated at a preselected velocity based upon rundown of the missile guidance set VCE integrator by two cartridge-actuated valves that shut off propellant flow to the booster engine.

LANCE MISSILE: GM-52C

FEED SYSTEM



The propellant-feed system consists essentially of two end-to-end cylindrical aluminum tanks having a common internal bulkhead: the forward tank containing the fuel and the aft tank the oxidizer.

During sustained flight the LANCE missile round maintains, essentially, a ballistic trajectory. Positive expulsion of propellant into the thrust chamber is maintained over the entire trajectory. Pistons, driven by pressure from the SPGG, provide such capability.

The SPGG is made up of booster and sustainer grains within an insulator case. Upon receiving an electrical signal to fire, a pyrogen igniter, incorporating a safe/arm device, initiates the SPGG. Gas pressure rapidly builds up to rupture the protective seals in the forward and intermediate bulkheads, and then pressurizes the upstream side of both pistons. When the pressure ruptures the protective seals in the fuel transfer tube and aft bulkhead, fuel and oxidizer begin to flow into the injector manifolds where they ignite hypergolically. Positive expulsion is maintained, with pressurization of propellants, throughout engine operation. Overpressurization is prevented by a hot-gas relief valve (HGRV) that maintains a nominal 1250 psi in the chamber forward of the pistons by venting excess gas overboard. A spin torque system uses SPGG gases which exhaust overboard through two spin nozzles for a controlled 1.5 seconds to provide positive spin torque

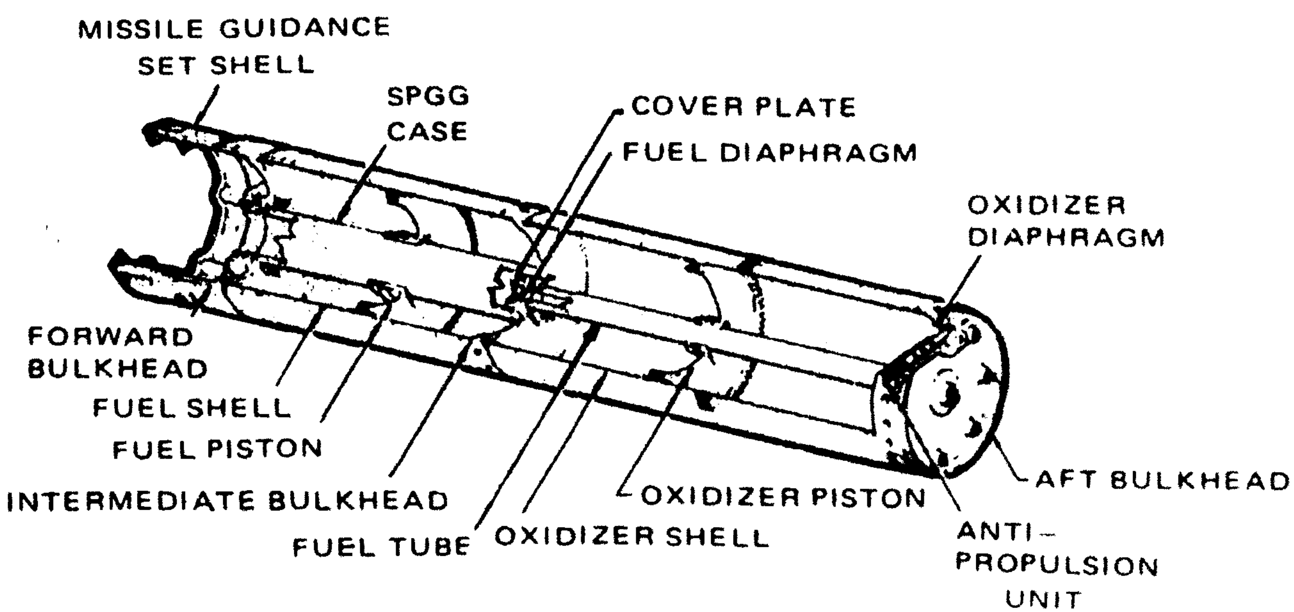
TANK BODY ASSEMBLY

The tank-body assembly includes a welded one-piece aluminum outer structure and the following subassemblies:

Aft bulkhead	Fuel tank shell
Fuel-transfer tube	Fuel piston
Oxidizer piston	Forward bulkhead
Oxidizer tank shell	Guidance set shell
Intermediate bulkhead	Control surfaces pads
SPGG case	Antipropulsion device

This assembly provides: (1) protection against radio-frequency interference and environmental effects, (2) good surface electrical conductivity, and (3) hermetic sealing. It houses the propellants and the components that initiate and sustain propellant pressurization and expulsion, and it affords structural support for, and/or access to, the following items:

Engine and boattail	Electrical harness
Warhead section	Electrical umbilical interface
Fin control surfaces	Safe & Arm igniter assembly
HGRV and spin tubes	Guidance set
Sighting-and-laying cover	Forward launch fitting



GUIDANCE SYSTEM

The AN/DJW-48 missile guidance set (MGS) is a self-contained guidance system comprising three subsystems:

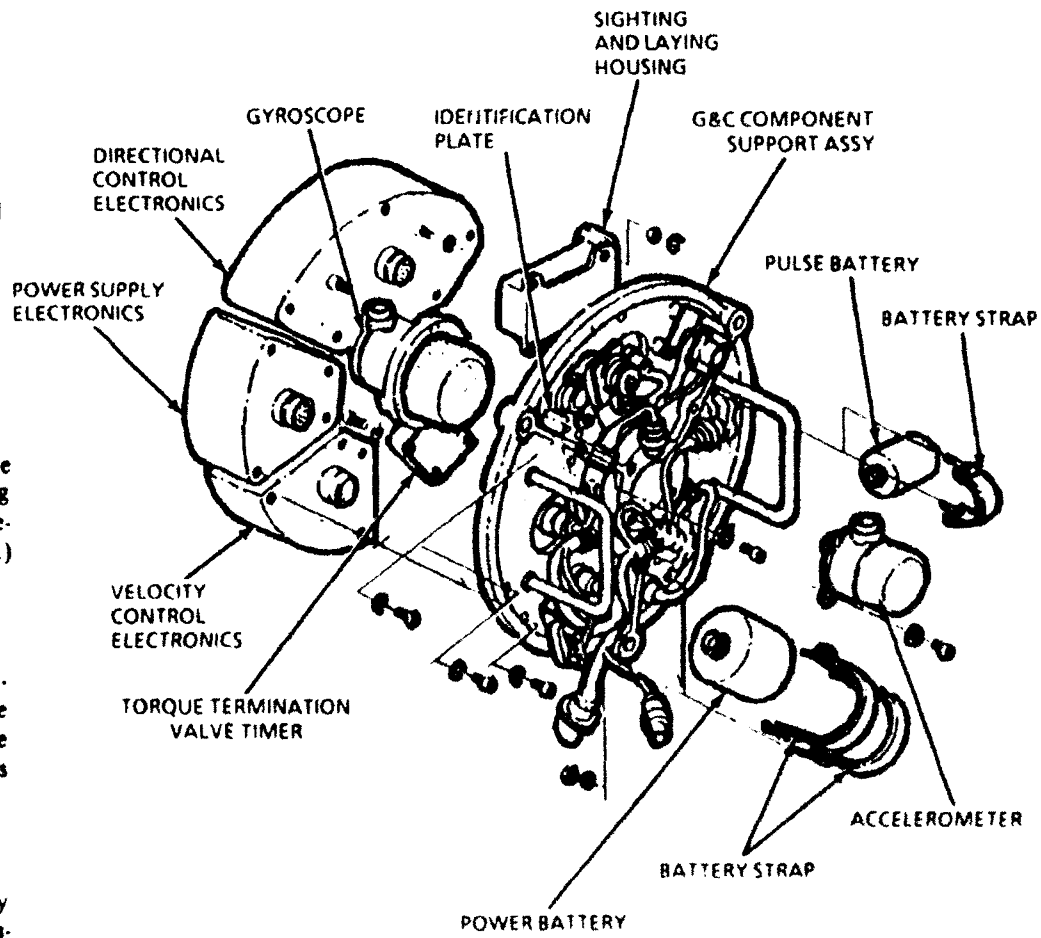
- Directional-control (DC) subsystem
- Velocity-control (VC) subsystem
- Power-supply subsystem

By regulating the four TVC valves, the DC subsystem maintains the correct missile round attitude about the pitch and yaw axes during the boost portion of flight. It consists, essentially, of a 2-degree-of-freedom gyroscope and the directional-control electronics (DCE) assembly.

The VC system, which contains an accelerometer and the velocity-control electronics (VCE) assembly, measures and monitors the missile round velocity and shuts off the booster portion of the engine at a preset value. After boost termination,the VCE subsystem controls the operation of the sustainer engine.

Missile round operating power is provided by the power-supply subsystem, which includes the power-supply electronics (PSE) assembly, a pulse battery, and a power battery.

The missile guidance set also includes a timer to fire the squibs that initiate closure of the torque-termination valve.



ACCELEROMETER

The Lance accelerometer senses acceleration in the missile longitudinal axis. The output of the accelerometer is used by the velocity system to provide the desired booster engine cutoff velocity and to equalize thrust/drag force variations in the sustain phase. The location of the accelerometer in the G&C package is shown on page 5.

The originally fielded Lance accelerometer (Systron-Donner Model 4841) incorporated a pivot and jewel bearing mechanical moving system in a sealed oil-filled vacuum enclosure. This accelerometer underwent a major recycle program during the 1978 timeframe. Recycling was required because the helium-nitrogen backfill gas mixture was leaking into the liquid filled moving system. At turn on, a gas bubble would form in the gas saturated liquid and attach itself to the moving system. The bubble created a buoyant force which caused the output voltage to drift. Xenon gas was used to replace the old gas mixture because it had a slower leak rate and was more soluble in the damping fluid.

Since testing of the fielded Model 4841 accelerometer indicated that its life would be limited to the 1992/1993 timeframe, a new accelerometer was developed and qualified. This new unit is a Sundstrand Q-flex design that senses acceleration through the use of a quartz disc which has been chemically etched to provide a movable circular inner-section supported by two quartz hinges. This Q-flex sensor assembly is air damped so that it does not have the bubble drift problem that contributes to the limited shelf-life of the Model 4841 accelerometer. Since the Q-flex accelerometer is relatively new, its electronics approach the state-of-the-art for this type of design. Therefore, the unit should have a field life of 25 years minimum.

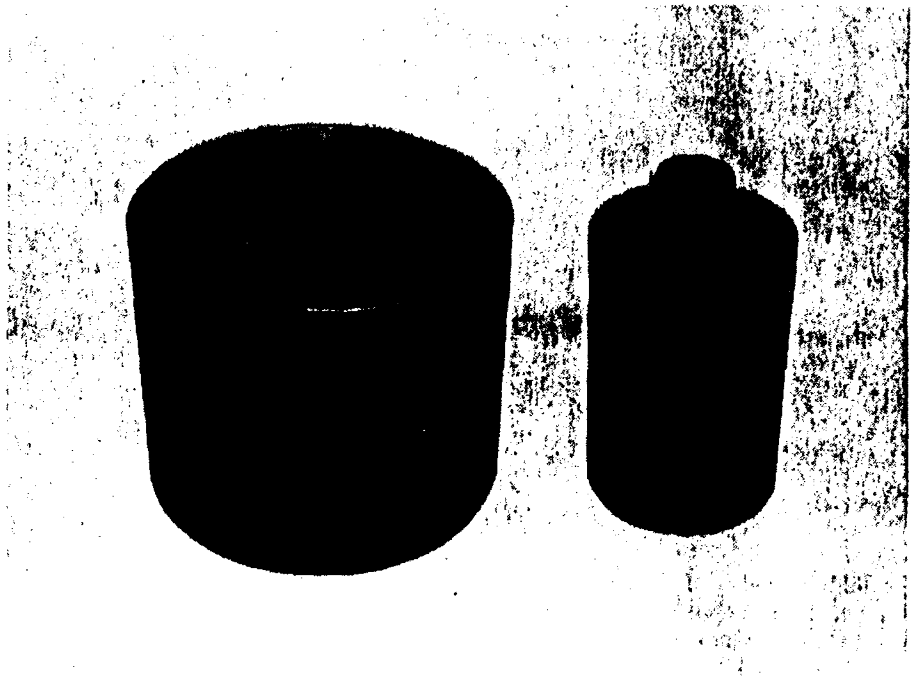
MODEL 4841 ACCELEROMETER

THERMAL BATTERIES

Two squib match-activated thermal batteries supply the primary power for the missile guidance system during flight. These two batteries (power and pulse) are squib activated during the firing sequence. A firing current of 2 amperes is required for the squibs in each battery. This firing current is supplied by the Ni-Cad battery in the Monitor Programmer. Further events in the firing sequence are restrained until the batteries reach their minimum voltage levels. These minimum voltage levels must be attained within 3 seconds after battery activation.

The power battery furnishes the input voltage for the power supply electronics and the thrust vector control valves. This 28-volt-nominal battery must supply a minimum of 22.5 volts into a resistive load of 13 ohms, with short-duration peak resistive loads of only 2 ohms, for the duration of the flight. The power battery weighs 2.75 pounds and is about 4-inches long and 4 inches in diameter.

The smaller 28-volt-nominal pulse battery supplies the inflight firing currents required to initiate the squibs of the boost termination and spin torque termination valves. These currents are as high as 22 amperes during short pulses for a period of 8.5 seconds after activation. The pulse battery weighs 1 pound and is about 3-inches long and 2 inches in diameter.

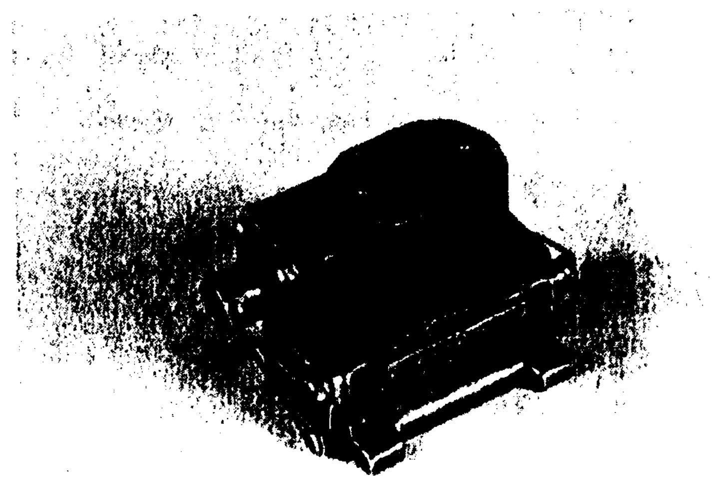


POWER BATTERY PULSE BATTERY

SERVOVALVE

The Lance servovalve is a two-stage hydraulic valve used to position the sustainer engine pintle to obtain the required level of thrust. The servovalve's first stage is a polarized electrical torque motor. Two 500-ohm coils are supplied with up to ± 10 milliamperes to proportionally position the second stage output spool. Proportionality is achieved through a feedback flapper attached to both the spool and the torque motor.

The servovalve operates at a UDMH supply pressure of 600 to 1200 psi during the missile's flight. Nominal flow through either of the two control ports is 0.55 gallons per minute per milliampere of command signal with 1200 psi at the supply port and less than 100 psi at the return port. The servovalve is designed to operate under the high shock and vibration environments encountered during the Lance booster engine operation.

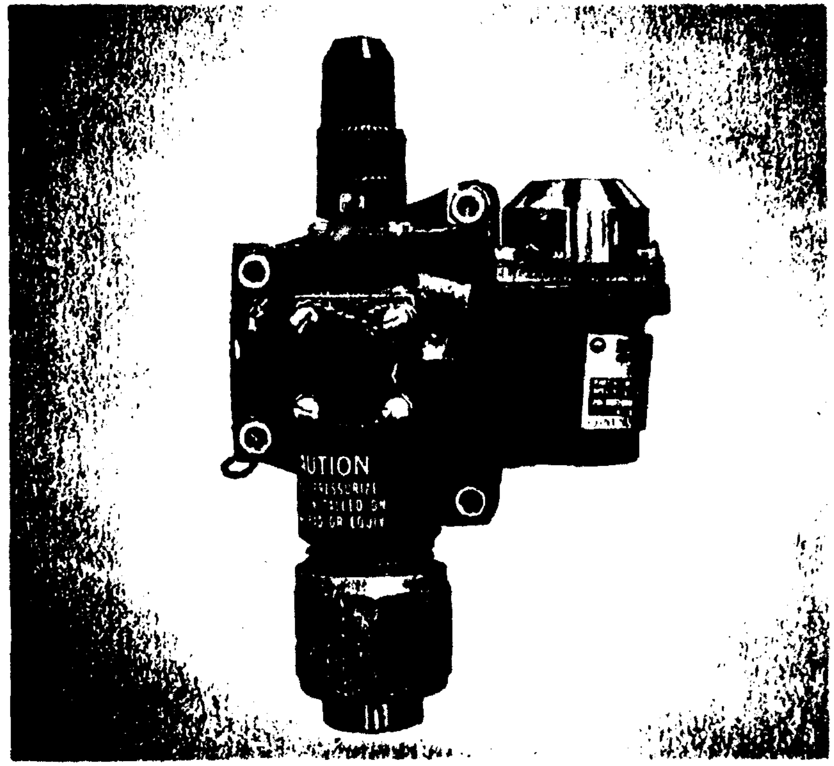


GYROSCOPE

The Lance gyroscope is a spring-wound, ball-bearing, squib fired inertial device that provides the reference for the boost phase flight control. The pickoffs are an "E-Core" magnetic-type which provide *angular* displacement errors in the yaw and pitch planes. The gyro bearings are high precision, low friction ball bearings. The gyroscope can only be activated one time. The location of the gyroscope in the G&C package is shown on page 5.

THRUST-VECTOR-CONTROL VALVE

The fuel injector manifold is tapped to fill lines leading to four TVC valves that provide directional control during booster operation. In response to *flight-path-heading error signals* sensed by the gyroscope in the guidance system these valves release streams of UDMH into the booster exit cone, thus producing a directional change in the thrust vector. The resultant side force causes the missile to move about its pitch or yaw axis for attitude correction.

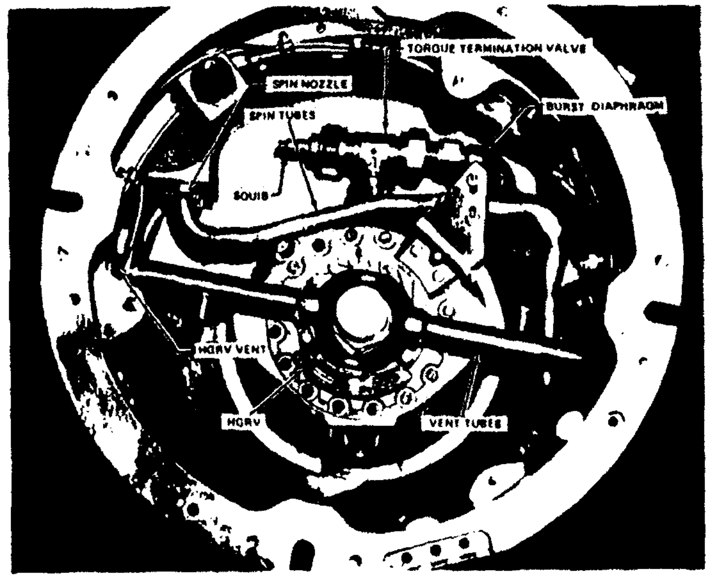


SPIN SYSTEM

The LANCE missile round spins throughout flight to provide aerodynamic stability and minimize target errors resulting from thrust malalignment. Spin-producing torque results from two sources.

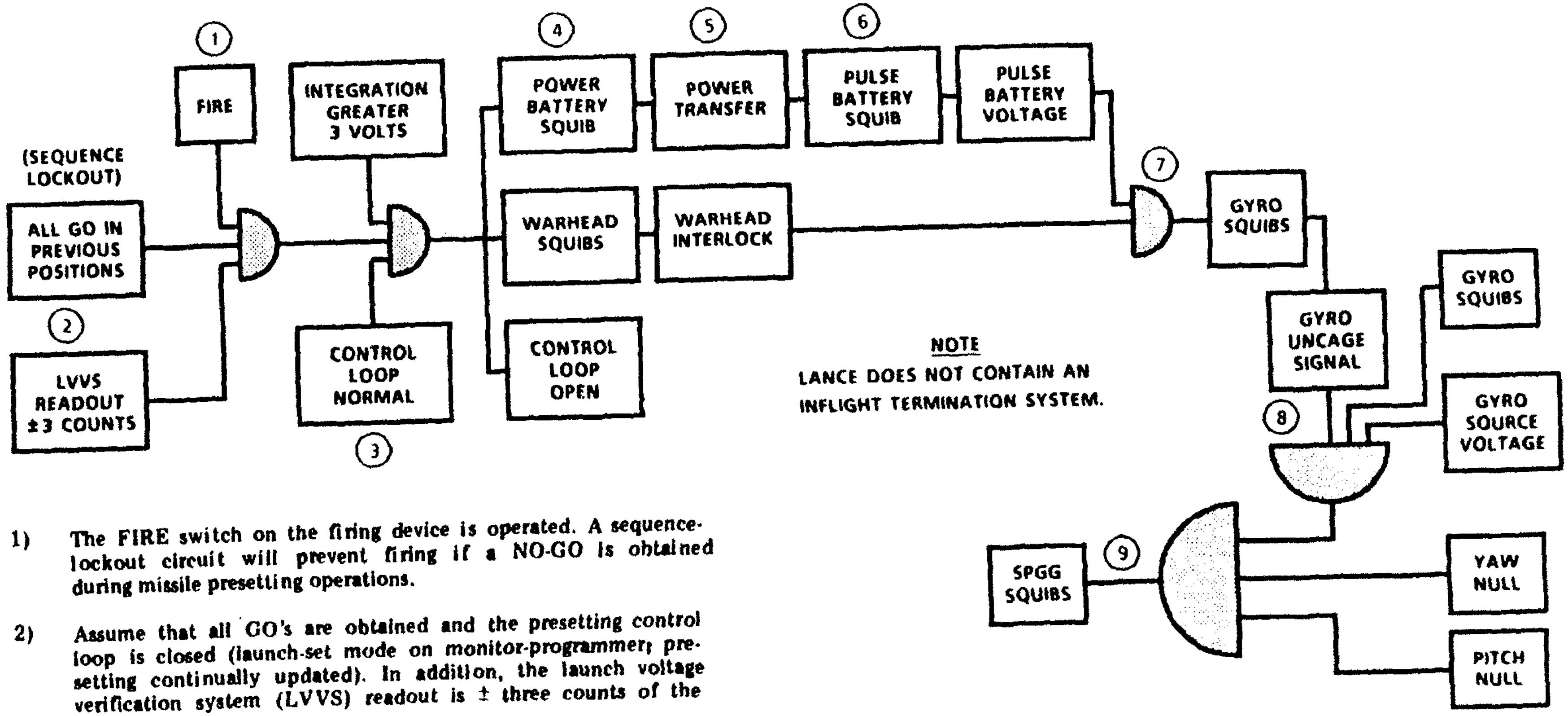
1) Initially, SPGG gas is vented through two diametrically opposed nozzles that are canted relative to the missile round's surface. The forces of the emerging gas effectively form a couple tangential to the surface, producing a torque about the longitudinal axis. This torque initiates missile round spin and is terminated by the timer-actuated closure of a torque-termination valve at nominally 1.5 seconds.

2) The trailing edges of the four control surfaces are deflected relative to the missile round longitudinal axis. Upon torque termination, the missile round velocity is sufficient to maintain lift-producing dynamic pressure on the control surfaces and, consequently, a spin-sustaining rolling moment on the missile round.



LANCE MISSILE: GM-52C

FIRING SEQUENCE



1) The FIRE switch on the firing device is operated. A sequence-lockout circuit will prevent firing if a NO-GO is obtained during missile presetting operations.

2) Assume that all GO's are obtained and the presetting control loop is closed (launch-set mode on monitor-programmer; presetting continually updated). In addition, the launch voltage verification system (LVVS) readout is ± three counts of the dial setting.

3) At the fire command, the control loop opens. This prevents any high current transients associated with squib firing from affecting the missile round presetting.

4) The fire relay fires the power battery squib.

5) After the power battery reaches its potential, power transfer occurs.

6) The fire relay fires the pulse battery squib.

7) Now, the gyro squibs are fired through the warhead section interlock, if the pulse battery voltage is proper.

8) Gyro firing produces: (a) the gyro uncage signal, and (b) gyro spin.

9) The next event is SPGG ignition if the following conditions are met.

The gyro is at zero. This is verified by checking the DCE yaw and pitch outputs to ensure that they are zero. However, the gyro signal would also be zero if the gyro power source were inoperative. Therefore, the gyro voltage source must also be verified.

GROUND SUPPORT VEHICLES

SELF-PROPELLED LAUNCHER (SPL) M752*

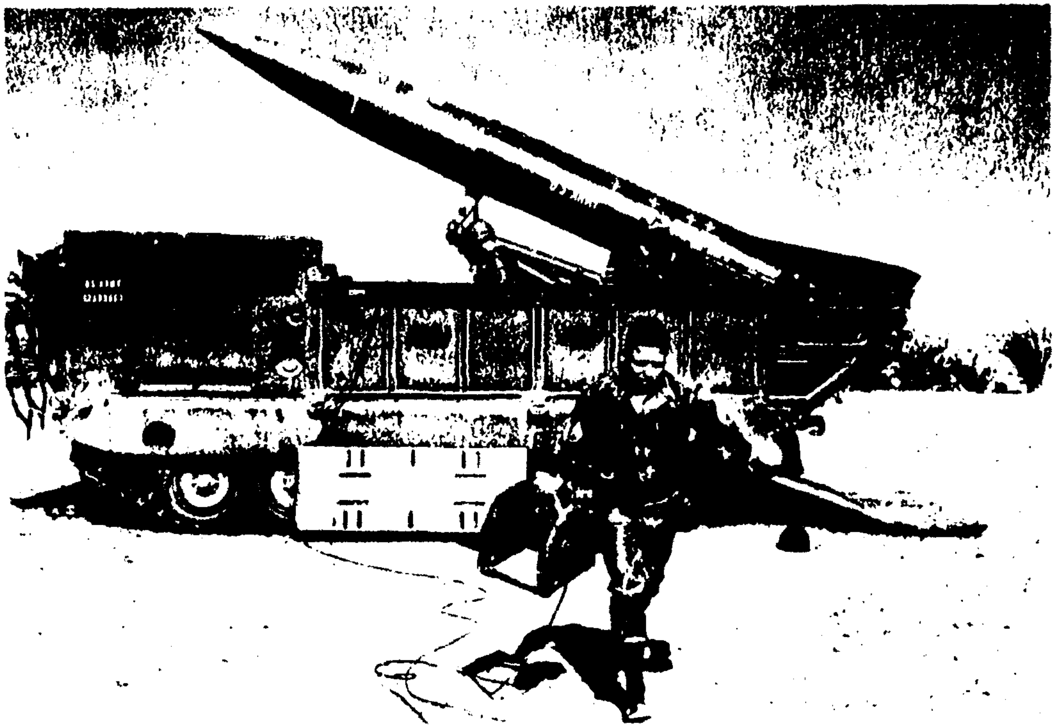
Combat weight **: 22,693 lb with heavy warhead section
22,198 lb with light warhead section

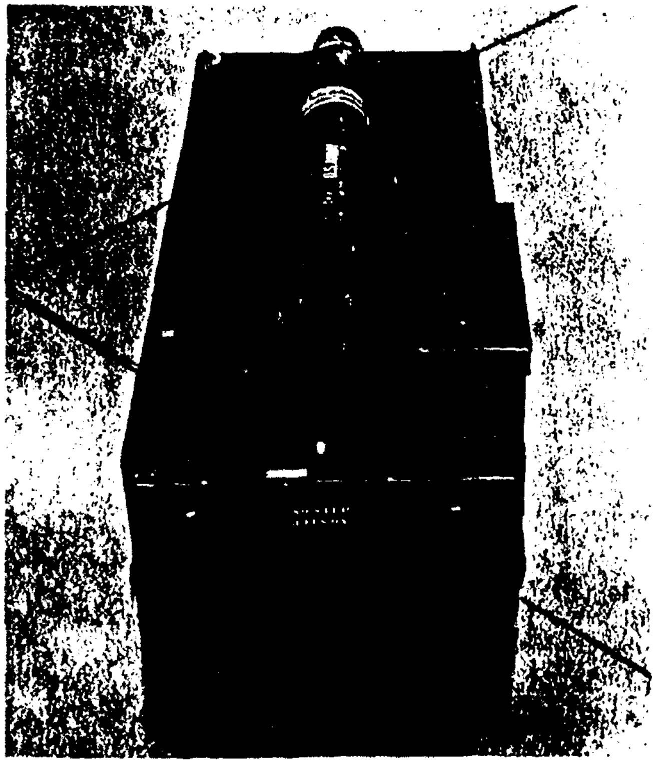
Length:	258.63 in.
Width:	106.75 in.
Height:	86.00 in. to cover 108.00 in. to top of raised cab
Power:	Model 6V53 diesel engine

Maximum sustainable highway speed: 40 mph

Swimming speed (still water): 3 to 6 mph

** Without crew

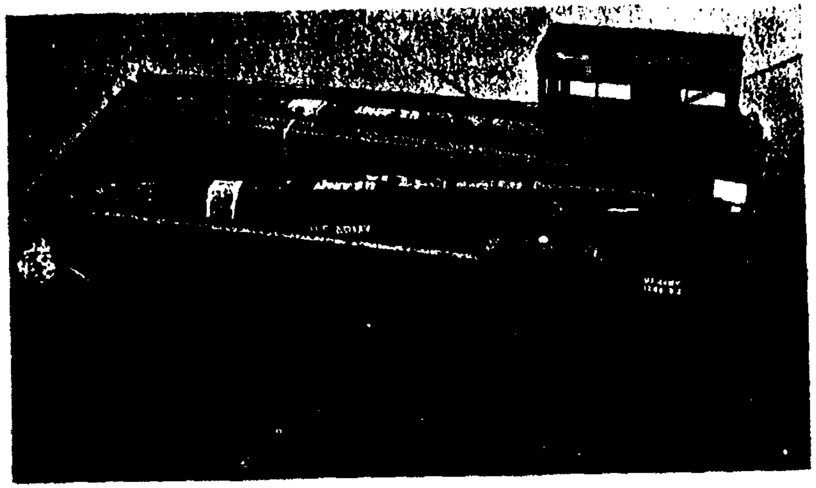




The LANCE self-propelled launcher (SPL) is a highly mobile, amphibious, tracked vehicle consisting of the M667 carrier and a basic launch fixture. The diesel-powered carrier employs the M113 power train, but is assembled in a hull peculiar to the LANCE system. Although the launch fixture functions as an integral part of the SPL, it can be easily removed and fitted with wheels, towbar, and stabilizing jacks to become a lightweight, towable launcher. Six members of the firing section can be accommodated aboard the SPL — four in the cargo area, and two in the tandem cab. Stowage brackets for on-vehicle equipment and miscellaneous crew equipment are provided on the basic carrier, along with a set of containers to house the control surfaces of the mated missile on board.

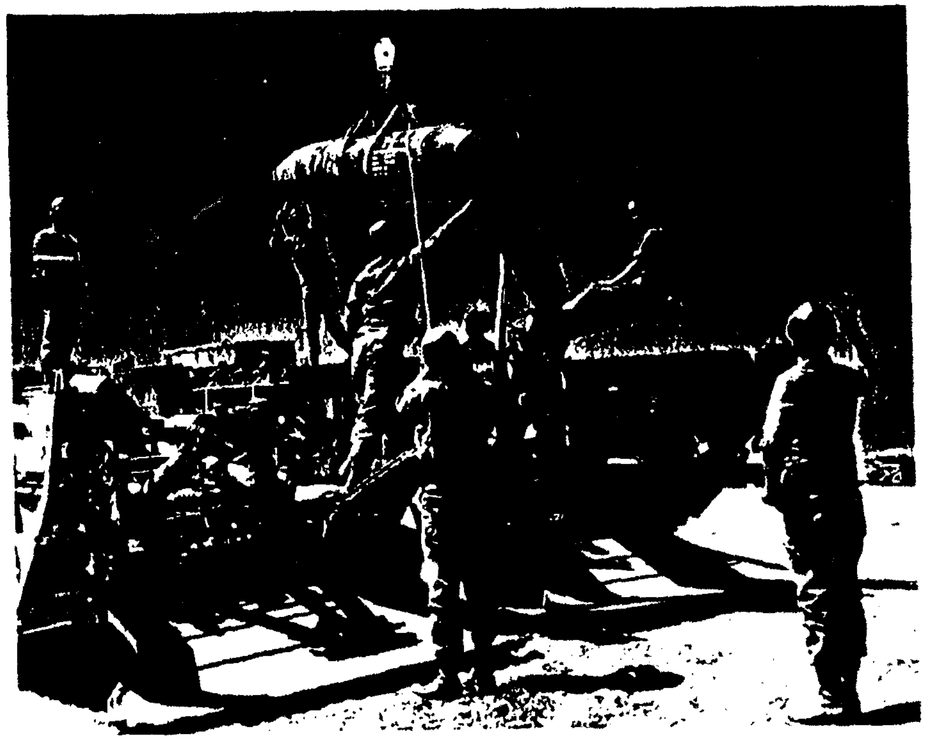
* The basic vehicle is managed by the U.S. Army Tank Automotive Command.

LOADER-TRANSPORTER (LT) M688A1*



Combat weight **:		Carrying two mated missiles, 25,746 lb with heavy warhead section 24,756 lb with light warhead section
Length:	258.63 in.	
Width:	106.75 in.	
Height:	86.00 in. to cover 108.00 in. to top of raised cab	

** Without crew



The primary function of the loader-transporter (LT) is to supply and load launchers. Like the SPL, the LT is based on the M667 carrier.

In the LT, a loader mechanism, supports and cradles for two mated missiles (or two main assemblages), and storage provisions for auxiliary equipment are mounted on the basic carrier. The manually operated loader mechanism is a constant-pressure, hydraulically powered boom crane with its base secured to the sponsons in the rear of the vehicle. Power is supplied through a power take-off gear driven by the vehicle's automatic transmission system which drives a hydraulic pump. The operator moves levers to control boom elevation and depression, 360-degree traversing, and reel-in/reel-out cable operation.

Auxiliary equipment stowed on the LT include a missile-handling sling, hand tools, vehicle cover, on-vehicle equipment, and crew equipment. In addition, two sets of control surfaces in containers are carried for the two mated missiles on board. The tandem cab accommodates the driver and an observer.

Both the LT and the SPL are air transportable and air droppable.

* The basic vehicle is managed by the U.S. Army Tank Automotive Command.

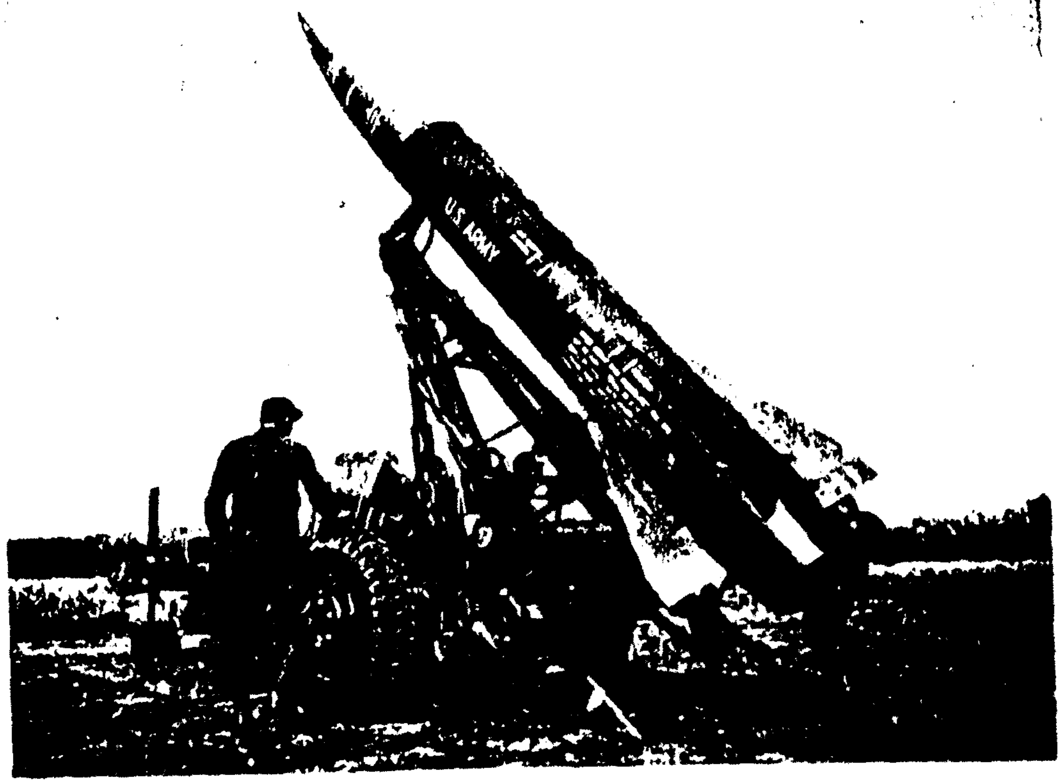
LAUNCHER ZERO LENGTH (LZL) M740

Weight:	Transport mode with fueled mated missile, 7,400 lb with heavy warhead section 6,880 lb with light warhead section
Length:	271 in. with missile 253 in. including towbar
Width:	78.0 in.
Height:	81.0 in. with mated missile 69 in. without mated missile

The LZL consists of the basic launch fixture and an adaption kit that include the stabilizing jacks, wheel and tire assemblies, trailing arms, towbar, and other items. Designed to be highly mobile, the LZL can be towed by any standard M35 type, 2½-ton vehicle at highway and cross-country speeds of 25 and 5 mph respectively. Over short distances, the LZL can be moved manually.

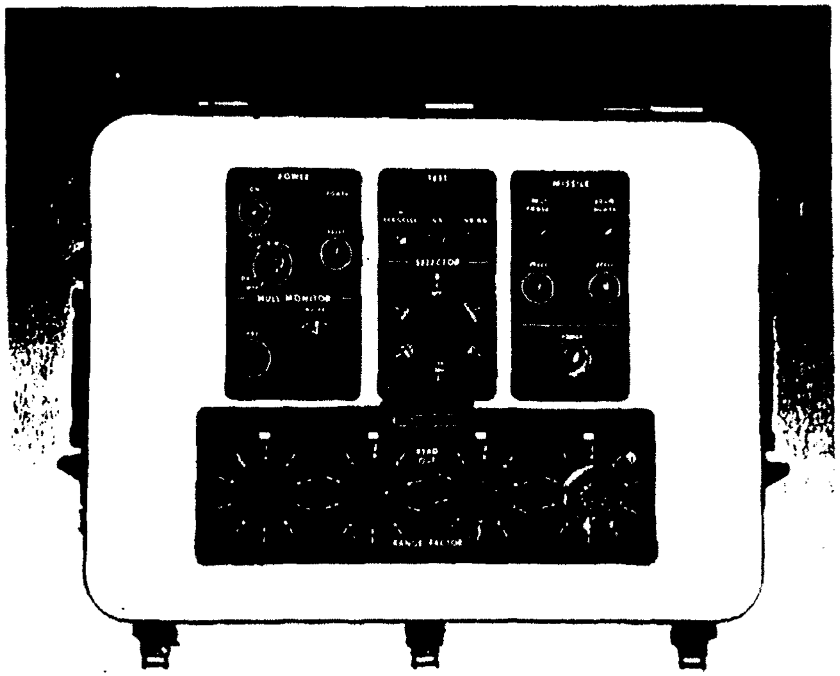
The basic launch fixture, which is common to the LZL and SPL, incorporates manually operated elevation and traverse mechanisms to aim the missile round. Prior to launch and for the first 5 inches of travel after ignition, the missile round is supported by two rear guide rails and a rotating forward support. After 5 inches of travel, the missile round leaves the rear guide rails and the forward support rotates out of the way, leaving the missile round in free flight.

The LZL may be air-transported by both the CH-47 helicopter (internally or externally) and the C-130, C-141, and C5A transport aircraft.



MONITOR-PROGRAMMER (MP) AN/GJM-24

Weight:	**105.00 lb**
Length:	**22.00 in.**
Width:	**18.00 in.**
Height:	**20.00 in.**



The MP transmits power to the *missile round during pre-launch* operations. It is also used to: (1) perform pre-launch monitoring; (2) insert range information into the guidance system before launch; (3) control the missile round safing, arming, and firing sequences; and (4) verify proper presetting to the missile round.

During pre-launch testing, the MP's GO/NO-GO monitor automatically provides a visual indication of the state of the missile round and prevents arming or firing if a NO-GO condition exists. The MP also monitors the arm circuit during pre-launch operations to prevent premature application of electrical power such as would occur in the event of a short circuit in the firing cable. If the arm circuit is prematurely energized, the ARMED indicator on the MP illuminates. In addition, a visual display of the velocity presetting is presented in the form of a visual readout.

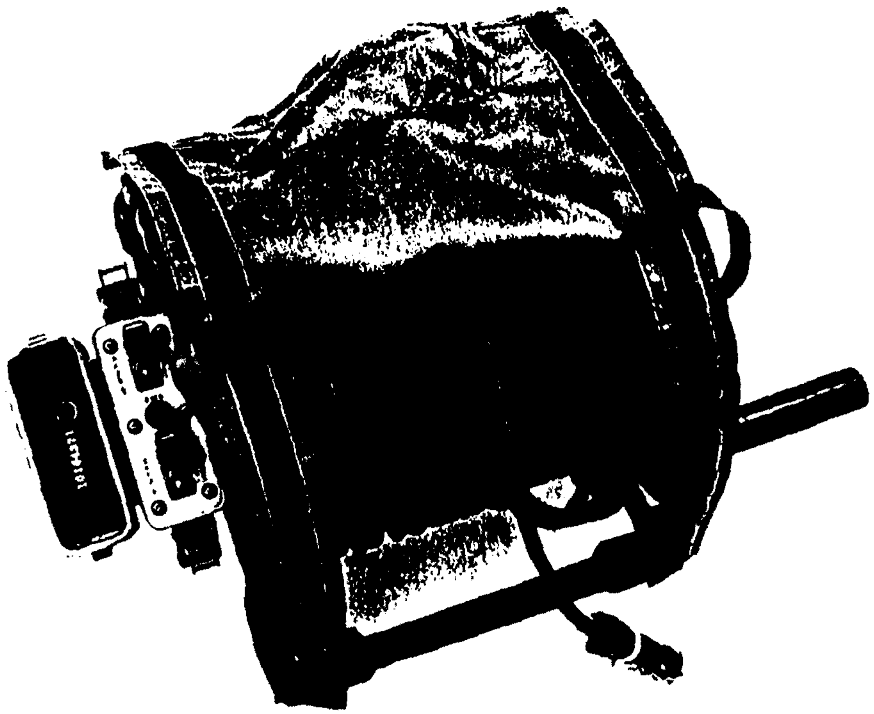
To control the launch phase of the firing sequence, the MP provides arm- and fire-interlock circuits that are activated by the ARM and FIRE switches on the firing device. Thus, if the missile round systems do not respond properly to the arm or fire command, the MP interrupts the firing sequence.

The MP, which is normally stowed on the forward right side of the launch fixture, is contained in a watertight case with external receptacles for power input, firing cable connections, missile umbilical connection, and MP maintenance testing. It receives power (24 vdc) from a 19-cell Ni-Cad battery. The controls and indicators on the panel face may be blackout illuminated for night operations.

FIRING DEVICE M91

Weight:	38.00 lb
Length:	16.00 in.
Width:	16.00 in.
Height:	14.00 in.

The firing device is used to arm and fire the missile round from a safe position 100 meters from the launcher. It consists of a firing cable, a handle assembly, a reel assembly with folding crank, and arming and firing controls. During pre-fire checkout, the firing device is connected to the monitor-programmer. Although it is usually mounted on the launch fixture, the firing device is portable.

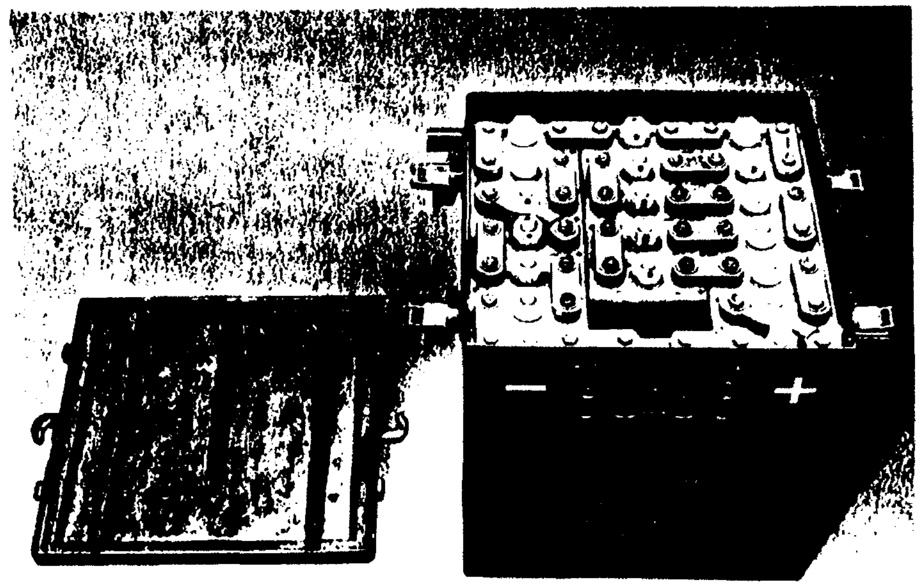


UMBILICAL CABLE

The two-piece umbilical cable is connected between the MP and the missile round, and is used to carry electrical signals from the MP to the missile round.

The short, expendable section has a lanyard-type disconnect at the missile round end and a cable connector at the other. The longer section, the launcher extension, has a receptacle for connection to the short section and a cable connector to plug into the MP. The launcher extension is secured to the launch fixture.

Weight:	80 lb
Length:	11.8 in.
Width:	10.5 in.
Height:	10.5 in.

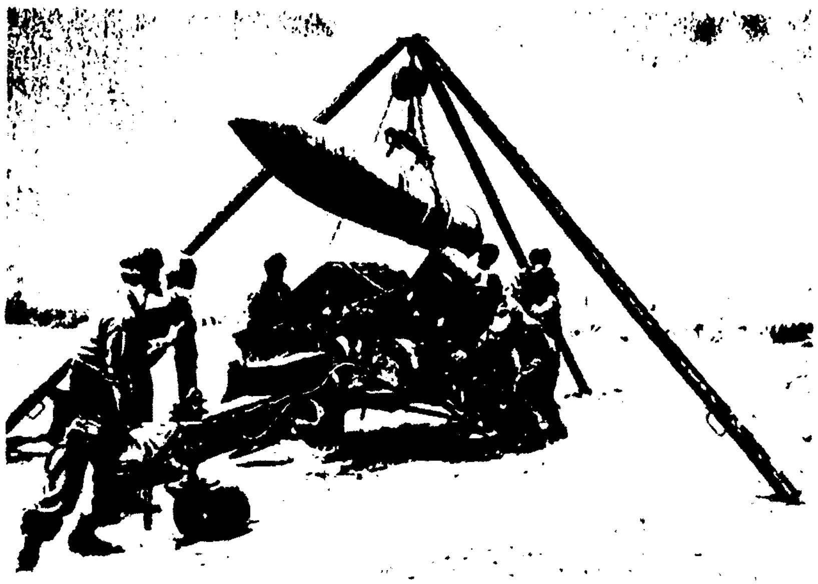


The primary ground power source for the LANCE missile rounds is a 19-cell, 24-volt, 34-ampere-hour Ni-Cad battery. This battery is housed in its own separate container on the launch fixture. A charging receptacle and a power outlet are mounted on the container so that the battery may be recharged without being removed from its container. The SPL's voltage regulator system may be used to supply recharging power.

HANDLING AND STORAGE EQUIPMENT

TRIPOD HOISTING UNIT
M38

Weight:	219.00 lb
Width:	271.00 in. (at base)
Height:	139.00 in.
Capacity:	4000 lb



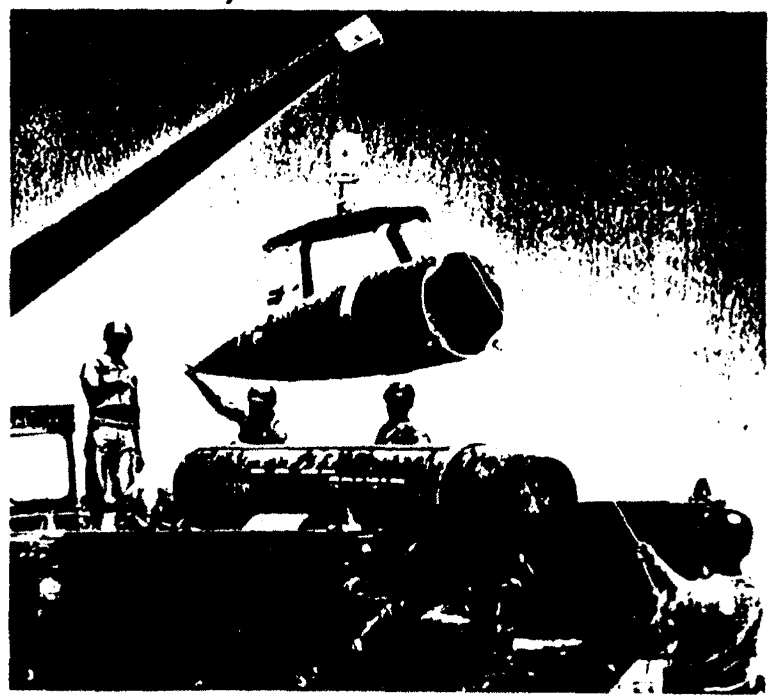
The tripod hoisting unit provides a means of reloading the LZL in the absence of the LT – for example, in airdrop or helicopter operations.

BEAM TYPE SLING
M22

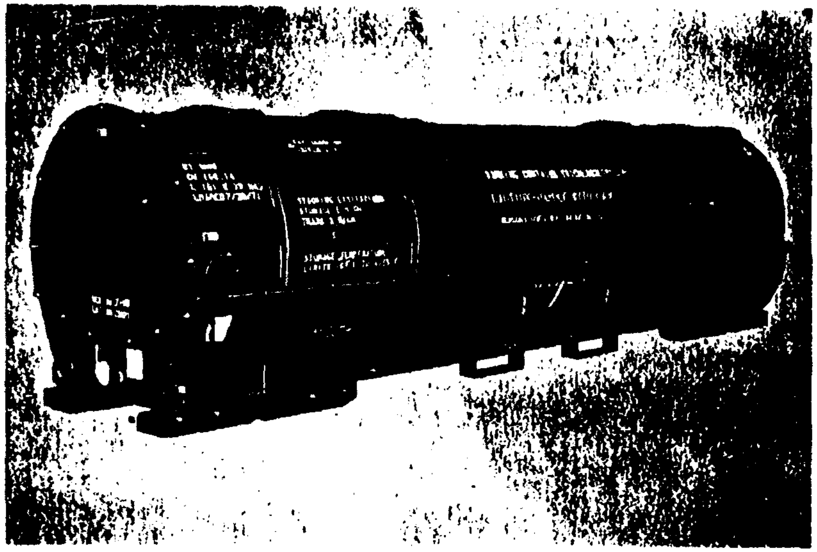
Capacity:	4200 lb
Weight:	65 lb
Length:	54.00 in.
Width:	1.25 in.
Height:	6 in.

The beam-type sling is used in: (1) loading or transferring the mated missile, main assemblage, or warhead section; (2) removing these items from their respective containers; and (3) mating the warhead section and main assemblage. Its cable assemblies are used for container and launch fixture handling.

The sling consists primarily of a 54-inch aluminum beam and two 4-inch-wide Dacron straps that are suspended from the beam with their centers 31 inches apart. Quick-disconnect fittings are provided on the straps for easy access.



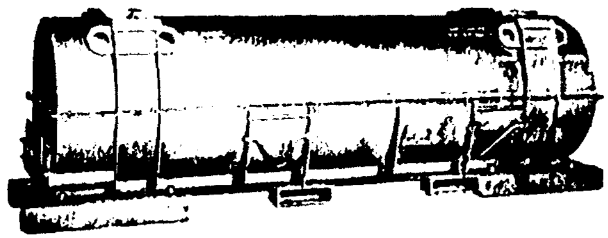
Weight:	1705 lb
Length:	160.50 in.
Width:	38.50 in.
Height:	42.50 in.



The main assemblage shipping and storage container is a stackable, enclosed container used for shipping and storing of LANCE main assemblages. It consists of steel constructed upper and lower shell assemblies and four hardwood skids. The lower shell assembly includes tiedown links, tee-head bolts for securing the upper shell assembly, two forklift channels, drain plugs, humidity indicators, and forward and rear polyethylene-padded cradle assemblies. The forward cradle assembly can be longitudinally adjusted to support a main assemblage or mated missile. Also included are desiccant, stowage provisions for the expendable short umbilical cable, a log-book container, and two *automatic/manual pressure relief valves.* Both valves operate automatically to equalize internal container pressure with ambient during, for example, air transport. Both valves also can be manually operated to equalize internal container pressbre before opening the container.

Weight:	1000 lb
Length:	116.00 in.
Width:	35.00 in.
Height:	38.00 in.

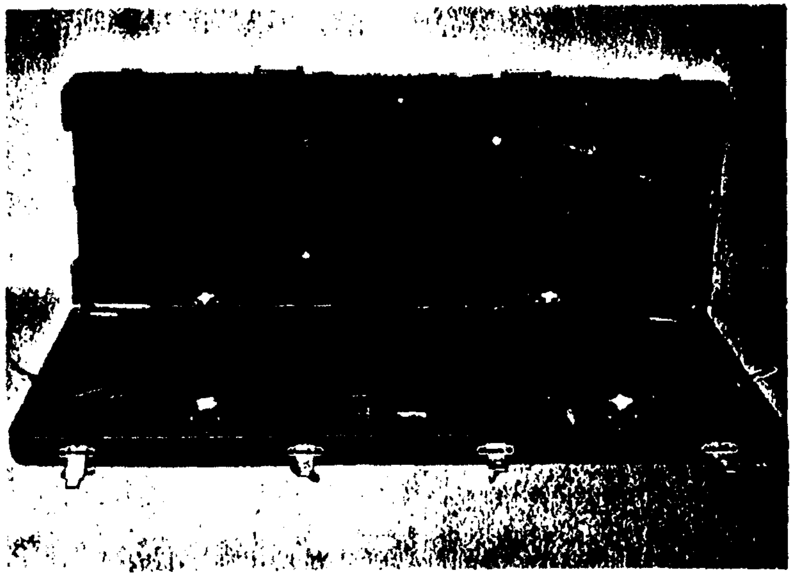
The warhead section container, which supports and encloses the warhead section during storage and transportation, rests on four wooden skids and supports the warhead section in a cradle assembly. During storage, the four warhead section T-bolts are secured to a support plate at the rear end of the container while the forward end of the warhead section is supported by a cradle and secured with a tiedown strap. Shock isolators cushion the cradle assembly. The container upper shell assembly is secured by T-bolts to the lower shell assembly, which is provided with lifting handles and removable drain plugs. A second container, external to the warhead section stowage area in the lower shell assembly, provides protection for log sheets and records.



Two-way, automatic/manual pressure-relief valves control the pressure within the container. Both valves operate automatically to maintain the internal-external pressure differential within allowable limits – during air transport, for example. By operating the valve pushbuttons, personnel can equalize internal and external pressures before opening the container. Desiccant is placed in the warhead section container to maintain low internal humidity when the container is sealed. A humidity indicator is provided to indicate desiccant saturation or container puncture or leakage.

HANDLING AND STORAGE EQUIPMENT

CONTROL SURFACE CONTAINERS M596 AND M597

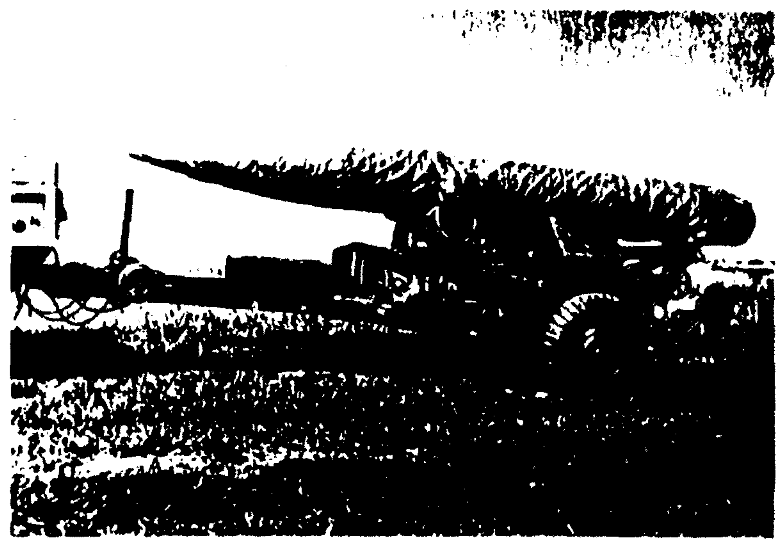


	Small Control Surface M596	Large Control Surface M597
Weight (lb)	63	77
Weight with control surfaces (lb)	91	111
Length (in.)	56.00	68.00
Width (in.)	28.00	28.00
Height (in.)	6.06	6.06

The four control surfaces are not installed on the mated missile or main assemblage during storage and transportation. They are housed in two containers (two control surfaces per container) lined with polyethylene to prevent shock damage to control surfaces. When the mated missile is mounted on the LZL, two control surface containers are secured to the launch fixture. In the LT or SPL, provisions for container stowage are built into the vehicle.

VEHICLE CARGO COVER

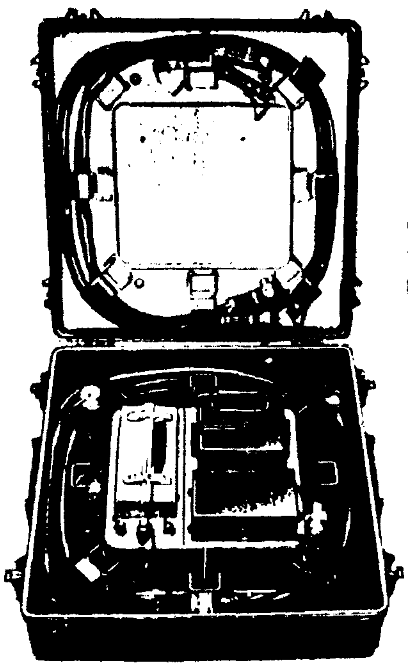
MISSILE COVER

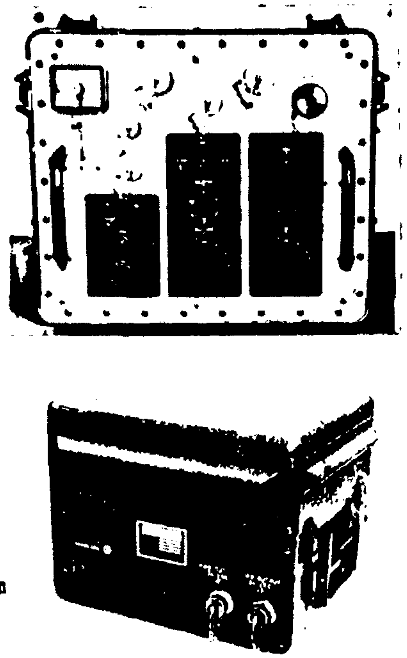


ELECTRICAL MAINTENANCE CHECKOUT EQUIPMENT

FIELD ARTILLERY MISSILE SYSTEM TEST SET AN/TSM 84 (GMSTS)

Electronic Test Set	Ni Cad Battery Assembly	Cable Storage Assembly
Weight: 100.0 lb	Weight: 91.0 lb	Weight: 107.0 lb
Length: 19.90 in.	Length: 14.50 in.	Length: 27.38 in.
Width: 22.75 in.	Width: 17.50 in.	Width: 27.75 in.
Height: 16.75 in.	Height: 12.75 in.	Height: 10.50 in.





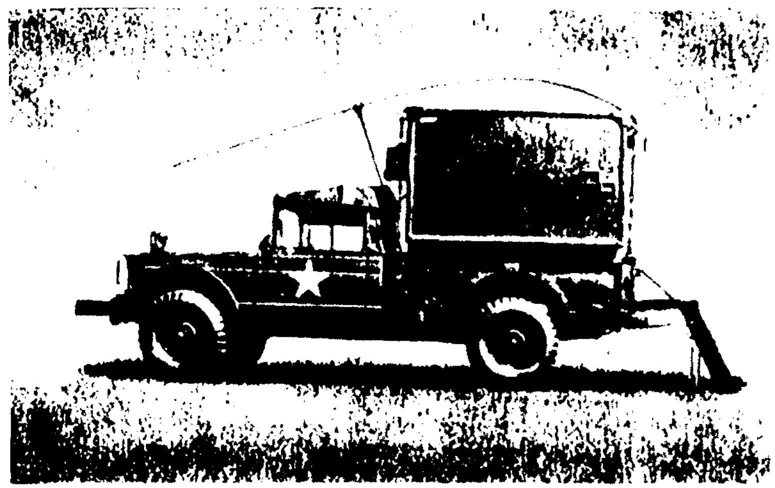
The guided missile system test set (GMSTS) is comprised of an electronic test set (ETS), a Ni-Cad battery assembly, and a cable storage assembly.

The ETS is used for surveillance and maintenance testing of the monitor programmer (MP) and the missile guidance set. It performs an electrical checkout of those circuits effecting the missile's prelaunch, launch, and inflight requirements. The performance of the item being tested is indicated by GO/NO-GO lights mounted on the ETS front panel. The ETS also checks the umbilical cabling system by fault isolation using its own umbilical cable.

The power source for the ETS is a 24-volt DC Ni-Cad battery assembly, which can also be used to power an MP.

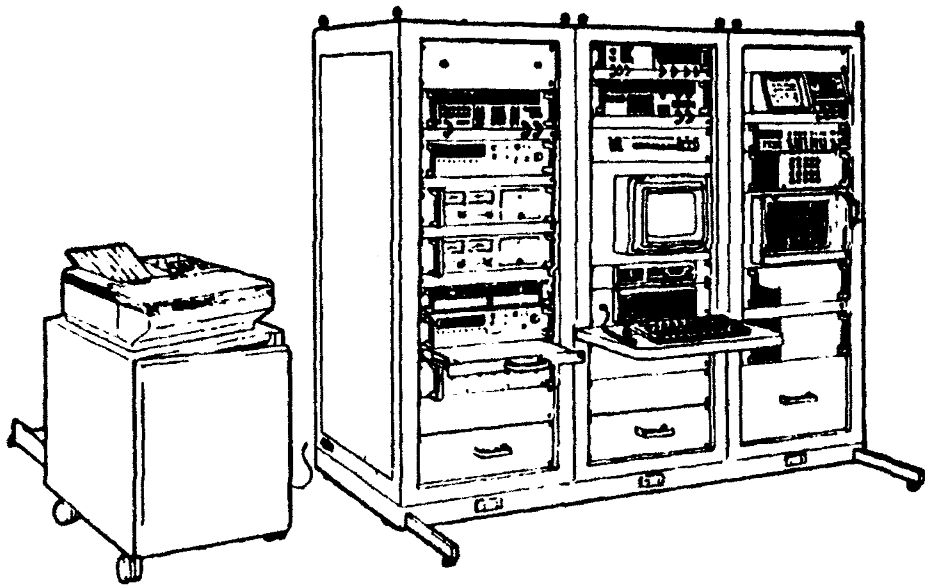
The cable storage assembly contains an igniter circuit tester, a multimeter, and a cable adapter assembly; all of which are used to check the rocket engine electrical system. In addition, the storage assembly contains all the cabling necessary to interconnect the ETS and the missile, or the ETS and the MP.

The electrical maintenance checkout equipment is housed in a modified S-250/G shelter, which is normally mounted on an M715 cargo truck, and can be mounted on an M561 truck. This configuration provides the LANCE contact maintenance team with shelter, radio communications, and mobility.



ELECTRICAL MAINTENANCE CHECKOUT EQUIPMENT

MP SPECIAL INSPECTION EQUIPMENT (SIE)



The SIE provides the capability for acceptance testing the Monitor Programmer (MP), the Circuit Card Assemblies (CCA) and the Power Supply. The SIE is fully automated with computer control of the programmable stimuli and measuring instruments and of the test sequencing. Routines are read into the computer memory from two discs. The slot 1 disc contains system and library routines. The slot 0 disc contains specific programs for the unit-under-test (UUT).

Operator instructions and test sequences are shown on a display monitor. Test results are dumped from the computer memory into a printer buffer to give a permanent printout record at the end of testing, independent of other operations of the computer. Therefore, other tests can be started while the printer is printing out previous tests results, since the latter operation is comparatively slow.

The UUT is connected to the SIE via an adapter that is specific to a UUT or to a related group of UUTs. Cable connections mate the UUT to the adapter, except for CCAs with edge connectors. These plug into receptacle slots on the adapter as do relays under test.

In addition to testing all MP UUTs, the MP SIE has been expanded by providing additional adapters and program discs to perform acceptance testing of the missile Power Supply Electronics (PSE), Directional Control Electronics (DCE), and Velocity Control Electronics (VCE), including their internal CCAs. An adapter is also available, as a tool, for testing a complete missile guidance set.

Other missile items can be tested with appropriate development of adapters and test programs using the existing console instruments.

PROPELLANT DRAINING KIT

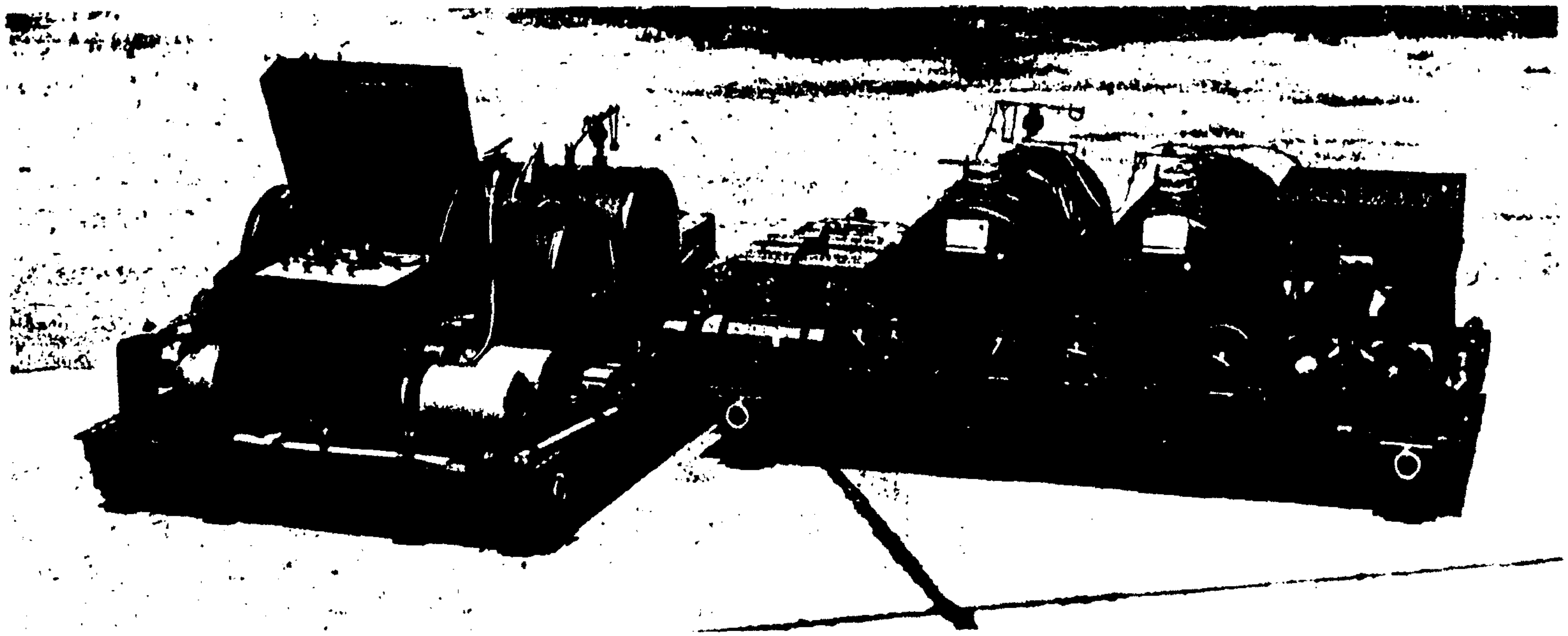
PALLET SIZE

Weight:	2400 lb
Length:	108.00 in.
Width:	66.00 in.
Height:	48.50 in.

LANCE is an inherently reliable and safe missile system. However, in the unlikely event of the tankage sustaining critical damage during handling, a propellant draining kit (PDK) is used in the field to safely drain the oxidizer (IRFNA) and the fuel (UDMH) from the LANCE M5 missile main assemblage (MMA) when the MMA cannot be removed to a draining facility.

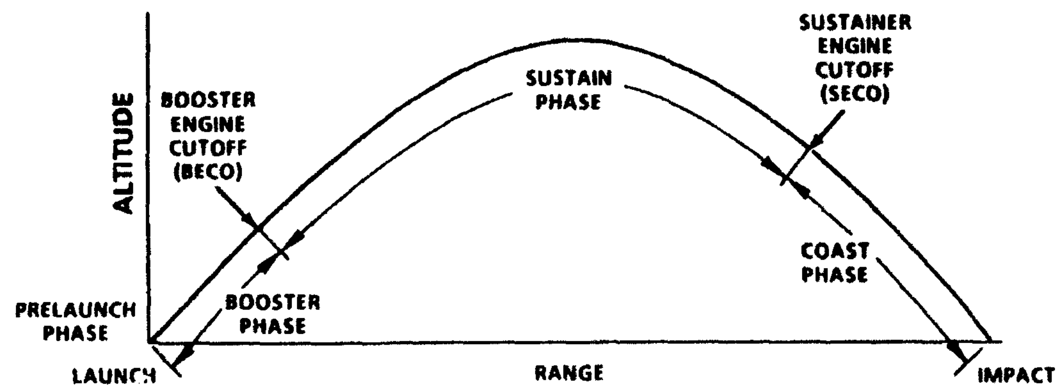
The complete PDK is comprised of an oxidizer draining kit (ODK) and a fuel draining kit (FDK). The ODK and FDK are mounted on separate channeled aluminum pallets which can be handled by forklift trucks and transported internally by truck, rail and aircraft, and externally from the Ch-47 helicopter.

Two techniques are used to drain a damaged MMA: pressure expulsion and pump transfer. A minimum crew of three men is required to safely perform the draining operation. Each crew member is equipped with protective clothing and an M20 breathing apparatus.



LANCE FLIGHT PERFORMANCE

PHASES OF FLIGHT



PRELAUNCH PHASE

- ACCURATE MISSILE SIGHTING & LAYING (AZIMUTH AND QE)
- PRESET BOOST TERMINATION VELOCITY (MP)

BOOST PHASE

- DIRECTIONAL CONTROL (TVC)
- ACCURATE END BOOST CUTOFF VELOCITY

SUSTAIN PHASE

- AUTOMET CONTROL (T = D)
- POSITIVE SUSTAINER ENGINE CUTOFF (SECO)

COAST PHASE

- MISSILE FREE BALLISTIC (NO CONTROL)

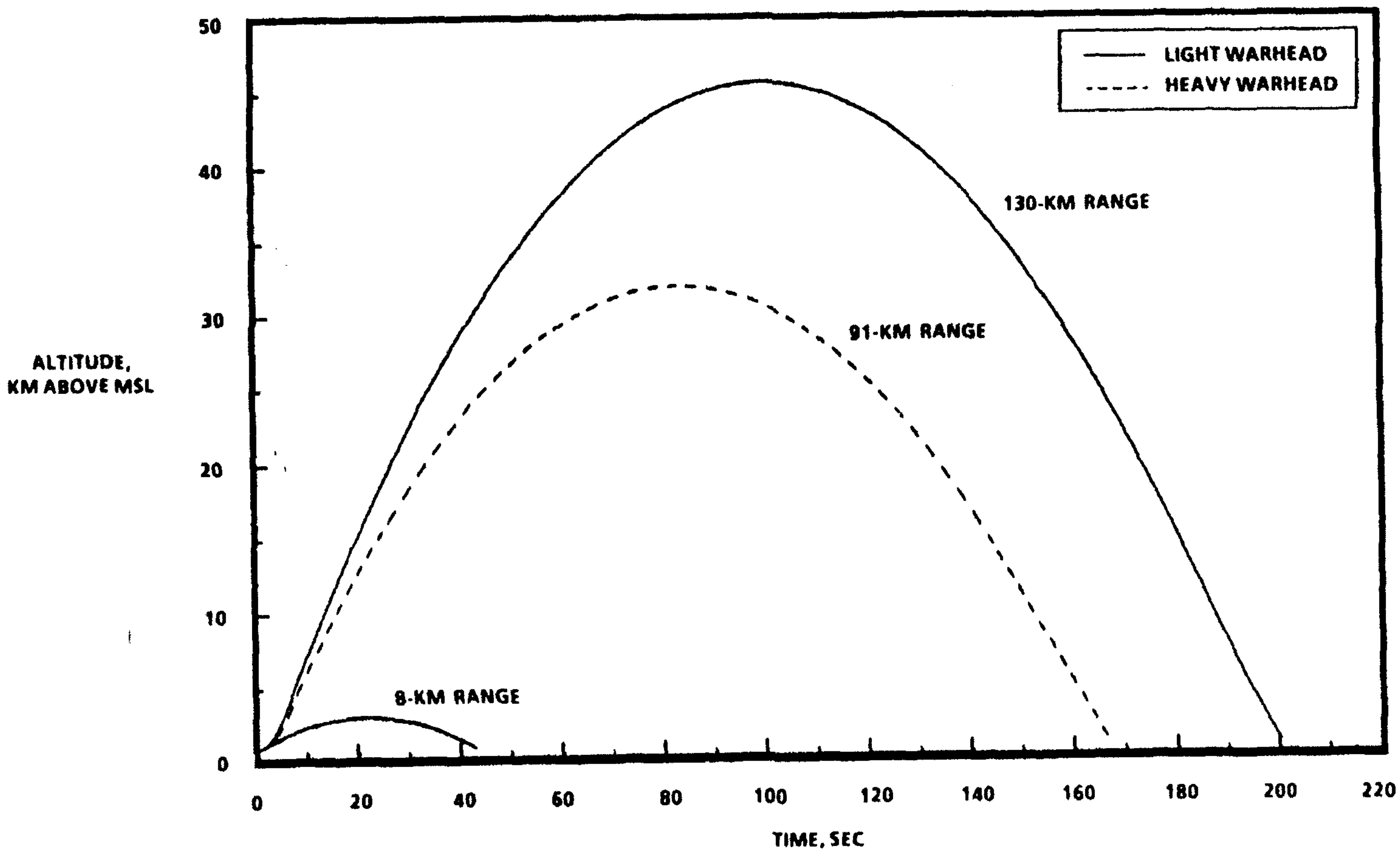
RANGE (KM)	W/H	QE (DEG)	BECO VEL (M/S)	APOGEE (KM)
8	HEAVY	48	273.01	3.123
8	LIGHT	48	274.18	3.148
91*	HEAVY	54	947.75	32.023
130*	LIGHT	54	1135.04	45.619

* FOR LAUNCHER ALTITUDE GREATER THAN 1000 METERS

LANCE FLIGHT PERFORMANCE

TRAJECTORY DATA

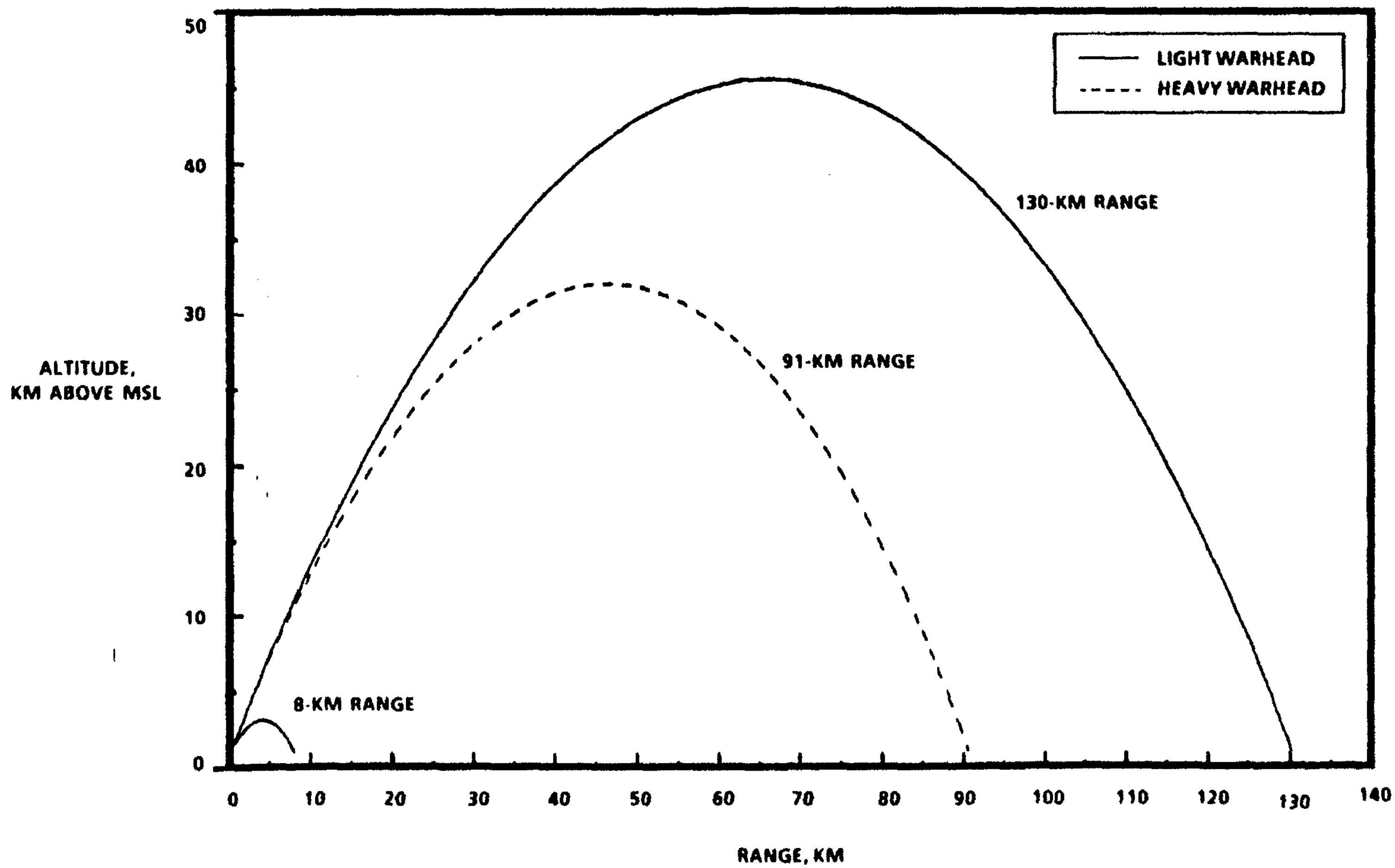
ALTITUDE VERSUS TIME
(LAUNCH ALTITUDE = 1001M)



LANCE FLIGHT PERFORMANCE

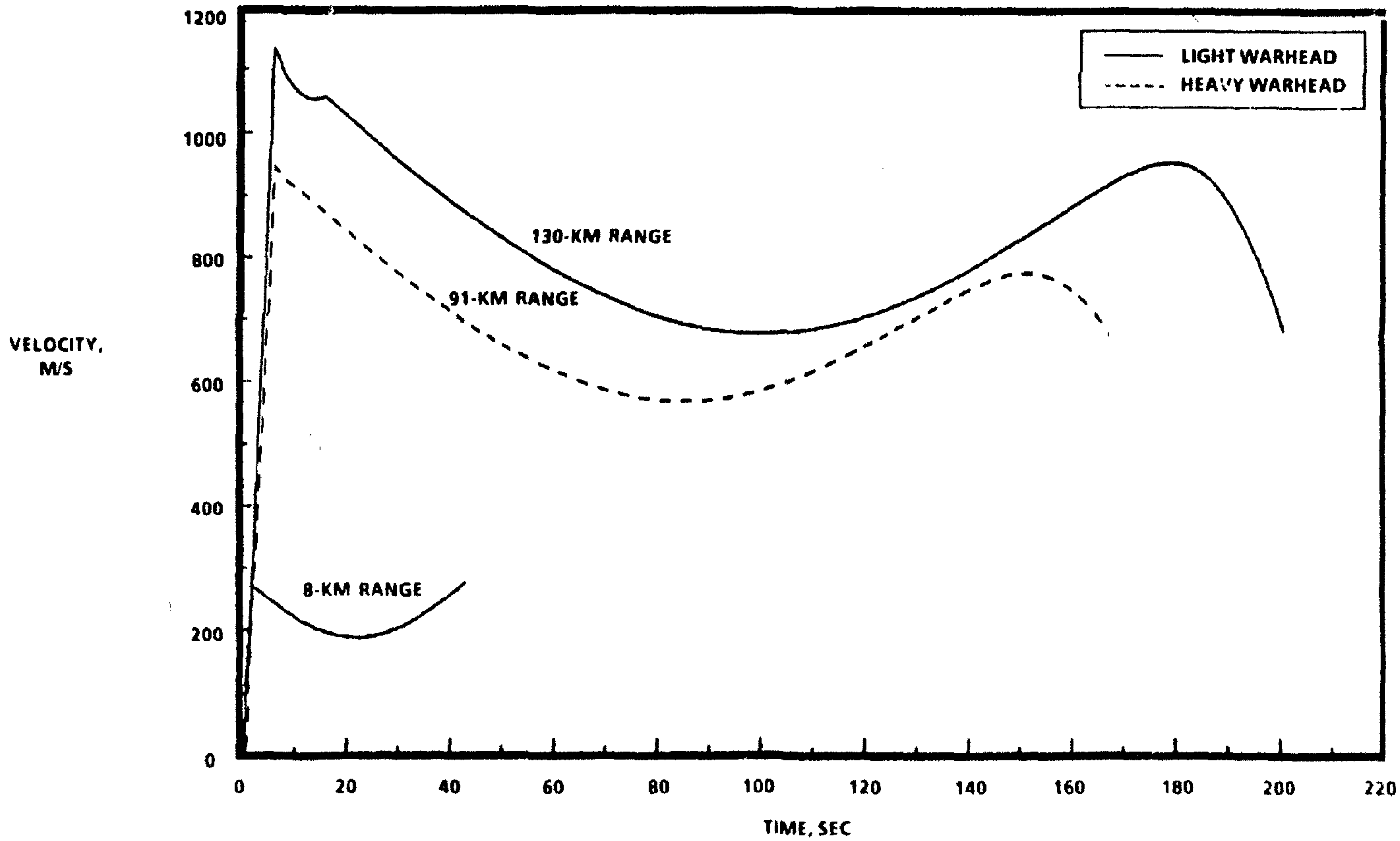
TRAJECTORY DATA

ALTITUDE VERSUS RANGE
(LAUNCH ALTITUDE = 1001M)



TRAJECTORY DATA

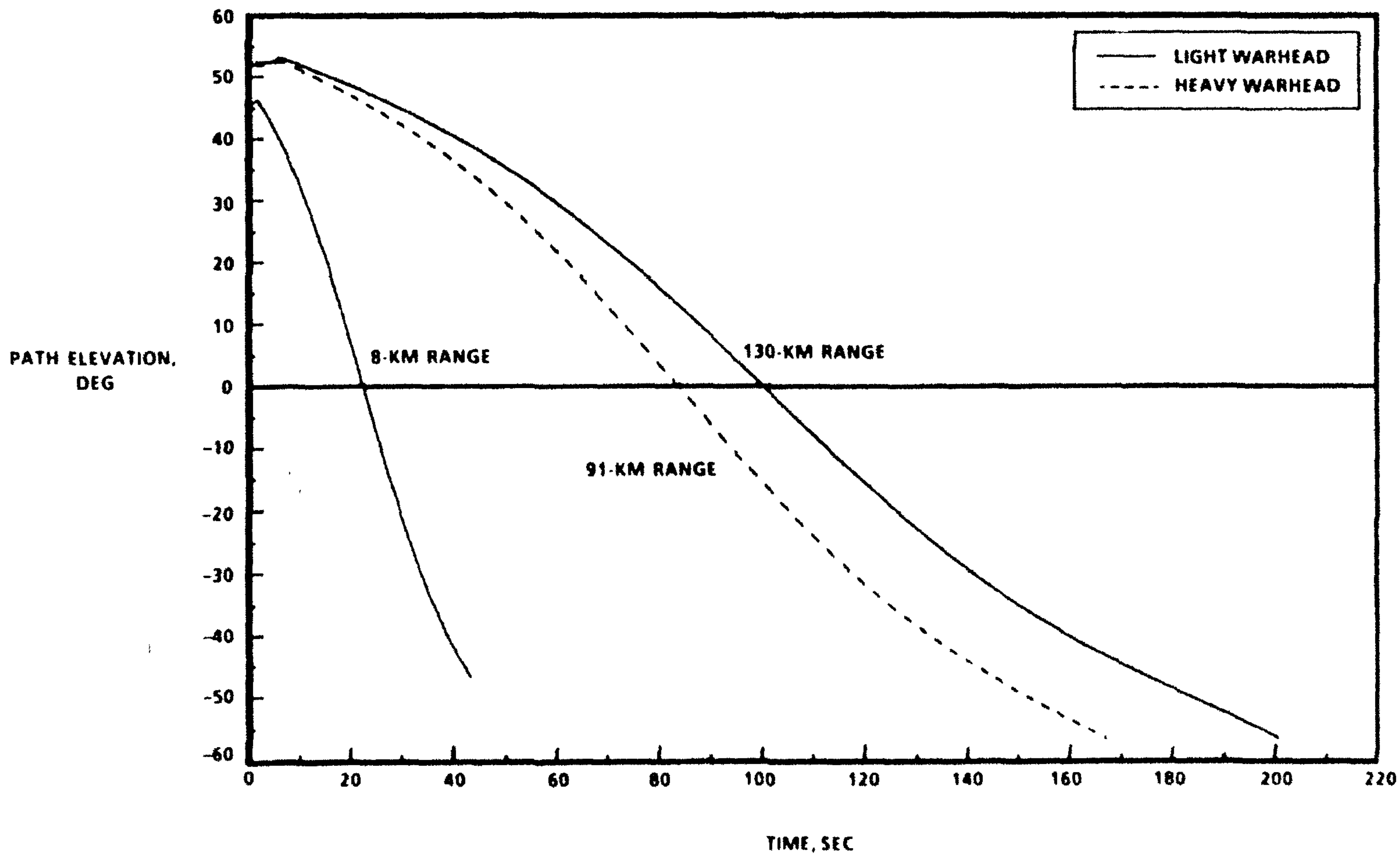
TOTAL VELOCITY VERSUS TIME (LAUNCH ALTITUDE = 1001M)



LANCE FLIGHT PERFORMANCE

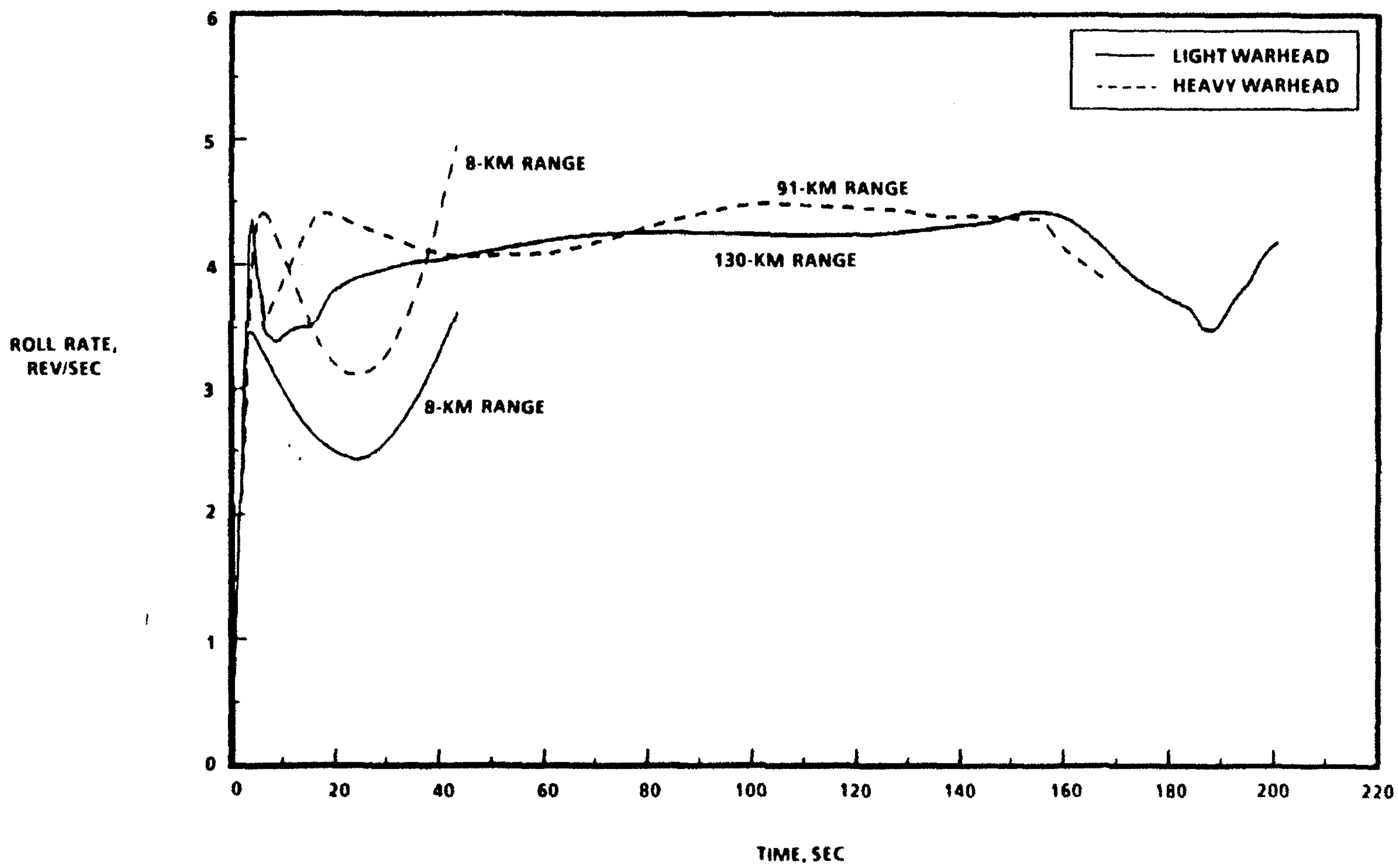
TRAJECTORY DATA

PATH ELEVATION VERSUS TIME
(LAUNCH ALTITUDE = 1001M)



TRAJECTORY DATA

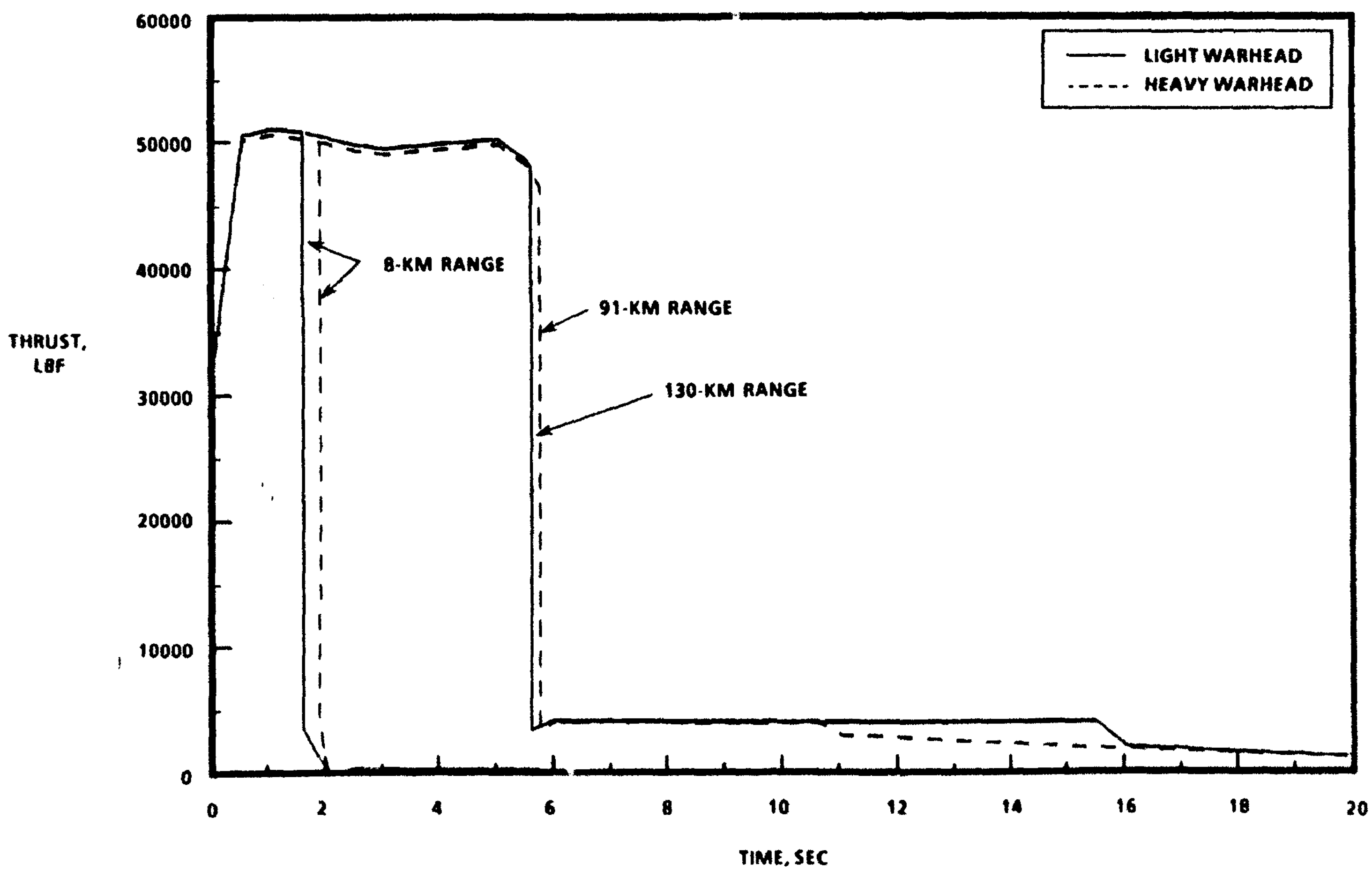
ROLL RATE VERSUS TIME (LAUNCH ALTITUDE = 1001M)



LANCE FLIGHT PERFORMANCE

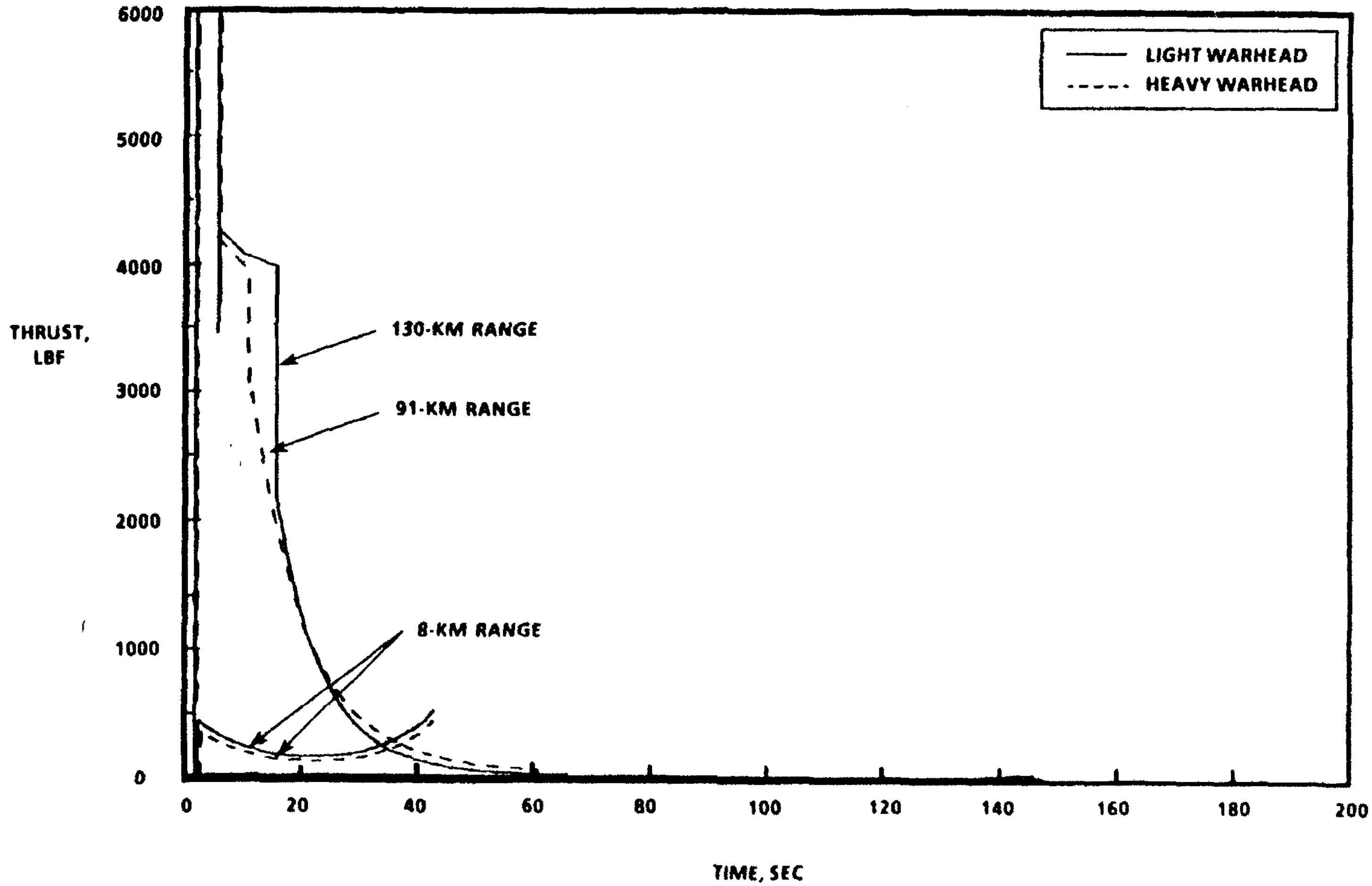
TRAJECTORY DATA

THRUST VERSUS TIME (BOOST PHASE)
(LAUNCH ALTITUDE = 1001M)



TRAJECTORY DATA

THRUST VERSUS TIME (SUSTAIN PHASE)
(LAUNCH ALTITUDE = 1001M)



APPENDIX C

STATEMENT OF WORK

FOR

CLUSTERED LANCE BOOSTER

FOLLOW-ON STUDIES

The Clustered Lance Booster (CLB) studies conducted under contract DASG60-92-C-0120 indicate that the concept is technically feasible. However, the studies also revealed that significant problems must be resolved and addressed before a development program can be defined.

The following statement of work identifies the tasks which must be performed/accomplished prior to entering into a development program. The tasks are presented by engineering discipline ie Design, Performance Analysis, Structures, Aerodynamics, etc.

I. <u>DESIGN</u>
The design activity will address two areas, the flight hardware and the ground support equipment (GSE).

A. <u>FLIGHT HARDWARE</u>

1. <u>CLUSTER DESIGN</u>
A redesign of the baseline will be required to incorporate the dynamic loads determined by this study. Of major concern, is attachment of the control surfaces which are projected to be seventy five percent larger then the current Lance design. The present tank body bulkhead is inadequate to absorb the loads which will be generated through these fins. This condition will require a redesign of the fin attach points and additionally therefore, the method used to secure the aft end of the cluster.

2. <u>INTERSTAGE DESIGN</u>
A predicted high angle of attack, plus thrust, drag and inertia, loads the front end of the cluster, missile station 100, to such degree that the interstage fairing must be redesigned as a structural load transfer member.

3. <u>ELECTRICAL INTERFACE BETWEEN STAGES</u>
Establish electrical interface requirements between the CLB, the intermediate stage and the payload. Includes signal routing, power requirements, power source and cabling.

4. <u>DESTRUCT SYSTEM</u>
Evaluate type of destruct package required to meet range safety requirements and determine optimum installation location.

B. GROUND SUPPORT EQUIPMENT

1. LAUNCHER DESIGN
A conceptual design of the launcher will also be initiated in the follow-on program. The launcher will be designed as a fixed site, fixed quadrant elevation device. The launcher design will incorporate the restraint devices necessary to hold the cluster assembly on the launcher until full boost is attained.

2. HANDLING AND ASSEMBLY EQUIPMENT
Identify the equipment required at the launch site to assemble the CLB and to handle the assembled CLB and its major components prior to assembly. This task establishes design requirements only, and does not include actual hardware design.

3. GROUND CHECKOUT EQUIPMENT
Establish the design requirements for equipment required to electrically checkout the CLB at the launch site and assembly area. This task does not include the design of the equipment but will include functional flow diagrams of the circuits to be tested.

II. STRUCTURES TECHNOLOGIES

A. LOADS AND DYNAMICS

1. MAXIMUM PREDICTED FLIGHT LOADS
Using a finite element model of the vehicle which represents lateral and axial stiffness distribution, determine maximum predicted flight loads.
Note: Stiffness and mass distribution of the forward part of the vehicle will be required from its manufacturer.

2. CORRECTIONS TO AERODYNAMIC COEFFICIENTS DUE TO VEHICLE FLEXIBILITY
Using the finite element vehicle model, determine corrections to the rigid body aerodynamics ($CL\alpha$, $CM\alpha$, CP location, etc.) at various values of dynamic pressure for use in the vehicle stability analysis.

3. FIN FLUTTER ANALYSIS
   A comprehensive fin flutter analysis will be performed on the larger fin design (approx. 75% greater area) to ensure an adequate flutter margin throughout the vehicle flight envelope. Both subsonic and supersonic conditions will be analyzed.

4. LAUNCH ANALYSIS
   Using the vehicle finite element model, determine maximum dynamic loads induced upon the vehicle and launcher due to the launch event. Also determine tip-off, clearances, and other dynamic parameters associated with launch.

5. STAGE SEPARATION ANALYSIS
   Determine by means of dynamic analysis the critical parameters (separation velocity, clearances, loads, etc.) associated with the stage separation event.

B. MECHANICAL SYSTEMS

1. STRUCTURAL ARRANGEMENT STUDIES, AIRFRAME
   Perform necessary analytical studies, trade studies and support design to achieve an optimum weight airframe structure. This includes structural loads analysis, selection of optimum materials and analysis of various structural arrangement for strength and stiffness. Analysis will use both computer and hand methods for fuselage and fins.

2. STRUCTURAL STUDIES OF LAUNCHER AND LAUNCH SYSTEM
   Perform necessary analytical studies and nodes to support preliminary design of a launcher and to translate launcher requirements into missile structure. Analysis will use both computer and hand methods.

3. ESTABLISHMENT OF VEHICLE AND LAUNCHER STRUCTURAL DESIGN CRITERIA
   Prepare a preliminary structural design criteria document for the missile and launcher structure. This document establishes all structural strength and stiffness requirements for both the missile and the launcher. It is a joint Loral/Army approved document which furnishes requirements for the final design phase.

III. SYSTEMS PERFORMANCE

A. IGNITION DETECTION SYSTEM

The purpose of the ignition detection system is to determine if each of the engines ignites. The vehicle would be constrained on the launcher until all the engines have ignited. Abort procedures would be initiated if any of the engines failed to ignite. The system would most likely utilize sensors to detect pressures essential for engine ignition. It may be possible to integrate the ignition detection system and the propellant depletion detection system required by the booster cutoff system. Trade studies will be conducted to define the optimum configuration for the system, and design requirements will be established for the selected system.

B. BOOST CUTOFF SYSTEM

Booster thrust must be terminated in a manner which will preclude asymmetrical thrust. One concept would use a timer which terminates thrust before the earliest predicted propellant depletion time. The timer would be inexpensive and reliable, but it would result in a loss of total impulse. A second concept would use pressure sensors to detect when propellant depletion begins in any one of the propulsion systems, at which time thrust would be terminated in the propulsion system undergoing propellant depletion and in the one diagonally opposite to it. It may be possible to integrate the propellant depletion detection system and the ignition detection system required during the launch sequence. Trade studies will be conducted to determine the optimum system for the CLB, and design requirements will be established for the selected system.

C. PROPULSION SYSTEM STATIC TEST AND FLIGHT TEST PROGRAM

Review CLB design requirements to determine which require verification by propulsion system static test and/or flight test. Prepare a coordinated test plan which will accomplish the desired design verification.

D. INVESTIGATE EFFECTS OF TEMPERATURE ON PROPULSION SYSTEM PERFORMANCE

Propulsion system performance in the feasibility study was based on ambient (59 $\pm$ 15 degrees F) static test data. After temperature requirements are established for the CLB, propulsion system performance at the temperature extremes will be determined by interpolating data from Lance temperature conditioned tests. (Lance static tests were conducted at -40 degrees F, ambient, and +140 degrees F.)

E. RANGE SAFETY STUDIES

The purpose of the study is to identify failure modes and their effects. Procedures to be followed in the event of failure will be recommended for each of the failure modes identified. Failure of one of the engines to ignite is of particular concern. The asymmetrical thrust of three engines would cause the vehicle to tumble if it were launched.

IV. AERODYNAMICS

A. AERODYNAMIC CONFIGURATION DEVELOPMENT

Aerodynamic analysis will be performed to optimize the Clustered Lance Booster (CLB) concept. The tail fin configuration and size will be determined to assure adequate stability and control characteristics throughout the flight envelope. In addition, upper stage nose shape and interstate fairings shall be chosen to minimize drag and enhance the missile performance. Aerodynamic loads (and load distributions) for the body and fins shall be generated for structural analysis purposes.
Specifically, the subtasks to be performed include:

1. Development of computer models to be used in stability, control and airloads analysis and the updating and revisions to these computer models as wind tunnel data becomes available.

2. Determination of the maximum aerodynamic loads (including the airload distributions) occurring on the missile body and fins.

3. Determination of the aerodynamic coefficients for the missile configuration to create an aerodynamic data set sufficient to perform 6-degree-of-freedom flight simulation and autopilot analysis.

4. Documentation shall be provided to support design reviews, inter-office communications and final report preparation.

B. WIND TUNNEL TESTING

Wind tunnel model testing will be required to refine the aerodynamic data base estimates generated under Task 1. The wind tunnel model shall have provisions for alternate tail fin designs, nose shapes and interstage fairings in order to optimize the overall missile configuration.
Specific subtasks to be performed include:

1. Establish the wind tunnel model and testing requirements to refine the theoretical aerodynamic data base generated under Task 1.

2. Provide wind tunnel model design information to support the preparation of detailed drawings required for model fabrication.

3. Provide a technical liaison between engineering and manufacturing during the model fabrication process.

4. Conduct the wind tunnel test to assure proper model configuration and quality of data.

5. Perform wind tunnel test data analysis to extract aerodynamic coefficients used in the aerodynamic data base.

6. Documentation of wind tunnel model requirements, test plans, wind tunnel data analysis report, and the preparation of inputs to the final report.

C. <u>AUTOPILOT DESIGN AND ANALYSIS</u>

Autopilot design and analysis will be performed in order to determine the optimum autopilot configuration for stability and control considerations. Requirements for an inertial measurements unit (IMU) and flight sensors will be determined and system components evaluated using vendor supplied data. Both modern and classical control concepts will be used in performing the autopilot analysis. Aerodynamic data will be initially obtained from the theoretical data base and updated as experimental data becomes available. Mass properties and missile flexibility data will be based upon estimates and continually revised as the design matures.
Specific subtasks to be performed include:

1. Evaluation of various autopilot concepts applicable to the Clustered Lance Booster design. Selection of baseline autopilot configuration.

2. Determine IMU and sensor requirements for the selected baseline autopilot configuration. Evaluate IMUs and sensors using data supplied by qualified vendors.

3. Optimize the autopilot baseline design by including the missile flexibility characteristics, by gain setting, by different filtering techniques, and by compensation.

4. Perform autopilot analysis to determine time histories, stability and response characteristics.

5. Documentation shall be provided to support design reviews, inter-office communications and final report preparation.

D. FLIGHT SIMULATION

Flight simulation will be conducted using a 6-degree-of-freedom (6-DOF) computer code that will provide a measure of performance based upon statistical (Monte-Carlo) techniques. Components are modeled, based upon their particular statistical characteristics (e.g., mean and standard deviation). Sensitivity analyses will also be performed using characteristics of the major components to determine their individual effects upon the missile performance. Aerodynamic data will be based on theoretical estimates and updated after experimental data becomes available. Other date required to support the flight simulation (mass properties, flexibility, and component characteristics) will be continually revised as the design matures.

Specific subtasks to be performed include:

1. Computer models will be developed to reflect the missile design to the maximum extent possible; to include aerodynamics, aeroelastic, autopilot, mass properties and system components.

2. A sensitivity analysis will be performed to determine component effects upon missile performance.

3. Performance analyses will be performed to determine the overall flight performance of the Clustered Lance Booster concept using statistical (Monte-Carlo) techniques.

4. Documentation shall be provided to support design reviews, inter-office communications and final report preparation.

E. BOOSTER CONTROL SYSTEM

The booster control system design and analysis effort will be directed toward the evaluation of different control concepts and the selection of one (or more, if required) for control of the booster throughout it's flight envelope. In addition, different guidance laws will be evaluated and control authority/response requirements will be quantified. Control system performance analysis will be conducted using control component characteristics obtained from qualified vendors.

Specific subtasks to be performed include:

1. Evaluation (and selection) of control system concept(s). The major control concepts to be considered include thrust vector control, aerodynamic control, and hot gas thrusters located

on the upper stage.

2. Establish control system authority and response requirements.

3. Evaluate various guidance laws and choose one for the Clustered Lance Booster concept.

4. Perform control system performance analysis. Characteristics for each control system will be furnished by qualified vendors.

5. Documentation shall be provided to support design reviews, inter-office communications and final report preparation.

F. <u>ELECTRONIC SYSTEMS DESIGN</u>

The electronic systems design effort shall be directed toward establishment of the guidance processor requirements (computational through-put, memory, control actuators, sensors, interfaces, etc.) and design based on the missile guidance, navigation and control/autopilot characteristics. Physical size and electrical power requirements shall be determined. Processors, electronic sensors, actuators and power sources shall be identified, evaluated and selected for the baseline Clustered Lance Booster concept.

Specific subtasks to be performed include:

1. Determine guidance processor requirements based on missile guidance, navigation, and control/autopilot characteristics.

2. Evaluate different guidance processor designs that meet requirements identified in (A) above.

3. Determine IMU, actuator, and sensor interface requirements.

4. Evaluate IMU, actuator, and sensor electronic/electrical characteristics using vendor supplied data.

5. Choose guidance processor, IMU, actuators, and sensors that meet required performance.

6. Documentation shall be provided to support design reviews, inter-office communications and final report preparation.

V. PROGRAM MANAGEMENT

A. PROGRAM MANAGER

Direct the activities of personnel assigned to the program, conduct in-house design reviews and periodically meet with the customer to review the status of the program.

B. FINAL REPORT AND DEVELOPMENT PLAN

Prepare a final report upon completion of all contractural tasks and prepare a plan defining a development program and the ROM costs associated with the development program.